NASA - GSFC MAGNETOSPHERIC CONSTELLATION MISSION DOCUMENT

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CONTENTS

PREFACE

This Document covers the following topics:

A.The Sun-Earth Connection Roadmap
B. Magnetospheric Constellation Mission Requirements
C. Mission Orbits
D. Mission Instruments
E. Nanosatellite Technology
F. Nanosatellite Subsystem Preliminary Study Implementation

Overall scope, architecture, writing, integration, subsystem technical review and consistency between sections, of this document, was my responsibility. Individual subsections were the result of excellent technical/written contributions of the many individuals associated with the GSFC Nano–Satellite/Micro-Satellite Technology Development Program. Acknowledgements and references are provided in the section under that title.

Al Lieberman

MISSION OVERVIEW

This document provides a summary of the work undertaken by the Micro/Nanosat Program to define an overall technical architecture for the MAGNETOSPERIC CONSTELLATION mission which is part of the Sun-Earth Connection Program. The launch date for this mission is 2010. The primary mission campaign objective is the Earth's Space Environment with a secondary mission campaign objective of providing Impacts of Space Weather. Some of the key technical capability requirements of the spacecraft platforms and their instruments are called out in the SUN-EARTH CONNECTION ROADMAP Strategic Planning for the Years 2000-2020 dated April 1997. This Roadmap places the development of New Technology at the forefront of Sun-Earth Connection Mission satisfaction in that it states up front on page 12: "To provide the key observations with minimal cost and risk requires developing important new technologies." Since a great deal of emphasis has been placed on new technologies, which are particularly applicable to this mission, by the roadmap (and also by the Micro/Nanosat Program) they are enumerated here:

Instrument Technologies
Miniature Instruments
Lightweight, miniaturized particles and fields instruments
 Magnetometer
 Plasma Analyzer
 Energetic Particle Detector
Miniature, sensitive analog electronics
Rad-hard durable instruments

Spacecraft Technologies
Miniature autonomous spacecraft
Satellite networks
Li-metal batteries
Advanced propulsion techniques
Micro-Avionics
Rad hard processors
Data acquisition from constellations

In addition, the "Mission Descriptions" portion of the Sun–Earth Connection Roadmap provides the overall mission scenario which was generally used to derive requirements during this pre-phase A study. This mission description was taken directly from the roadmap and can be found in figure 1

MISSION REQUIREMENTS AND ASSUMPTIONS

As noted in the section above, Figure 1 provides the overall mission scenario upon which more detailed mission requirements were derived. Various realistic tradeoffs were made based on the items enumerated above. Figure 2 provides the detailed set of requirements which have emerged from the pre–phase A study. The mission technical architecture outlined subsequently in this document is derived from these requirements.

Figure 2

Magnetospheric Constellation Mission

Preliminary Requirements
Mission Scenario 1
Revised 3-4-99 (by PVP)

Size	30 cm (12”) diameter x 10 cm (4”) height
Shape	Cylindrical disk
Volume	. 007 m^3 = 7,070 cm^3
Number of Probes	Minimum: 44 (2 satellites per orbit) Maximum: 104 (4 satellites per orbit) Initial satellite spacing within each orbit TBD
Launch Vehicle	Delta II 7925A
Mission Orbits	All perigees = 3 Earth Radii (Re) [Min. case] 22 highly elliptical orbits (2 satellites per orbit) Apogees = 10 to 52 Re in 2 Re increments [Max. case] 26 highly elliptical orbits (4 satellites per orbit) Apogees = 10 to 60 Re in 2 Re increments
Mission Lifetime	2 years
Radiation Environment	100 krads total dose over mission lifetime Latchup immune SEU = TBD LET = 90 (TBR)
Deployment	Line of apsides parallel to Sun-Earth line Perigee between Sun & Earth
Inclination	Typically 7.5° from Earth equator initially Initial Minimum: 0° from Earth equator Initial Maximum: 15° from Earth equator *Inclination angle expected to change as much as 40° due to lunar and other perturbations over the life of mission.*

Orbit Periods	0.97 days for Ra = 10 Re 10.4 days for Ra = 60 Re
Orbit Position Control	+/- 0.5 Re at each apogee → (0.833% of apogee radius)
Orbit Position Knowledge	Science requirement: ±20 Km Communications requirement: ± 60 Km
Eclipse Duration	Less than 1 hour (greater if orbit plane is in ecliptic) Duration maximizes twice a year near orbit equinox
Instruments	Electron detector Ion (proton) detector Magnetometer
Mass	10 kg (includes orbit insertion and attitude control fuel)
Power	4.5 watts continuous, generated by body mounted solar
cells	
	Batteries for eclipse periods

Power and Mass Budget

	Watts	**Kg**
Power System (includes harnesses)	1.00	2.50
Magnetometer	0.20	0.30
Ion/Electron Analyzer	1.00	1.00
Instrument Data Electronics	0.30	0.20
C&DH (includes ACS, propulsion)	0.80	0.25
ACS	0.30	0.45
RF Communications	0.50	0.50
Thermal	0.00	0.25
Propulsion propellant	0.005	1.25
Structure	0.00	2.50
Margin	0.395	0.80
Total	**4.50**	**10.0**

Battery	1000 cycle lifetime:	7 amp-hrs at 3.3 volts 30% max depth of discharge
	700 cycle lifetime:	2 amp-hrs at 3.3 volts 50% max depth of discharge
Power Bus Voltage	3.3 V +/- 5% regulated	
Thermal	[passive] insulation and coatings only [active] mini capillary pumped loop (mini-CPL)	
Stabilization	Initial spin = 40 to 60 RPM from launch vehicle Final spacecraft spin >= 20 RPM with axis normal to ecliptic	
Attitude Control	Spin stabilized by cold gas micro-thruster 20 rpm = 1/3 rev/sec Spin rate knowledge: < $2x10^{-5}$ rad/sec Spin axis position knowledge: < 0.1°	

	Spin axis drift rate: < 0.1° over 30 days
Orbital Maneuvers	[10 Re apogee] deltaV = 3000 N-s [60 Re apogee] deltaV = 7000 N-s
Sun Synchronization	Sun sensor Step rate = 2000 Hz (TBR) Resolution = 0.1°
Inertia	Izz/Ixx > 1.05
Instruments Data Rate	1.6 kbps total for all instruments
Overhead Data Rate	400 bits/sec for encoding, housekeeping
Recorder Data Rate	2 kbps
Data Storage	SSR (memory stack in multi-chip module) Maximum: 1792 Mbits for 60 Re apogee orbit (249 hours storage)
Transmission Rate	Up to 100 kbps
Command Rate	1 kbps
Communications:	
Transmit Power	500mw = -3.0 dBw (augmented by battery power)
Antenna	patch, monopole, or quadrafilar helix
Antenna Gain	0 dB
EIRP	500mw = -3.0 dBw = +27 dBm
Tlm Data Rate	32 kbps up to 250 kbps
Tlm Frequency	8470 MHz requested (X-band) Lower preferred as long as antenna does not violate inertia requirement
Cmd Frequency	7209.125 Mhz (X-band)
Ffwd/Frtn	749/880 for Category A mission at X band, 8400-8500 MHz
Ground Terminal	11m antenna
G/T	35.4 dB/°K X band
Quantity	1 for every 25 nanosats, near equator (Puerto Rico, Hawaii, Guam?)
Orbit determination	One-way doppler using USO Stability = $5x10^{-8}$ per day (tone ranging not in baseline)

As a result of the derived requirements of figure 2 and the subsequent trades to satisfy these requirements a set of assumptions was made on the mission technical architecture in order to make the mission a reality. These assumptions can be found in Figure 3.

Figure 3

<u>Nanosat Fundamental Assumptions</u>

March 24, 1999

A. Prior technology demonstration flights will help provide the confidence, and thus reduce risk, in autonomous operations.

B. Onboard watchdog timer within the C&DH unit will provide a master reset capability for microprocessor software hangups.

C. Nanosats beyond the first STP Constellation mission will be further sophisticated in inter-satellite communications.

D. All onboard instruments will operate at a predetermined constant output data rate. There will be no capability to vary instrument data rates feeding the data recorder system.

E. The average power used by the propulsion system for attitude control will be nearly zero. The assumption is either a battery is charged, or storage capacitors are used to activate the propulsion control valves.

F. The only systems operational immediately after deployment will be those essential in performing the delta-V maneuver to obtain the nanosat's final orbit. After the orbital maneuver is complete, normal spacecraft operations will commence, which includes turning on science instruments.

G. The entire constellation of nano-satellites will be identical except for the sizing of the orbital maneuver solid motor. For the highest apogee orbit nanosats, it is anticipated that the initial weight may exceed 10 kg., however after achieving on-orbit operations the net weight of all nanosats will be under 10 kg.

MISSION ORBITS

Introduction

The Magnetospheric Constellation Mission, which employs Nanosats, has the goal of placing multiple clusters of very small earth orbiting spacecraft into varying eccentric orbits to enable the science of the mission. NASA's Goddard Space Flight Center (GSFC) is sponsoring a Nanosat concept as a means of spurring technology advancement in disciplines related to spacecraft development. The primary goal is to produce a 10 kg spacecraft capable of boosting itself to the mission orbit and performing a science gathering mission for two years. A summary of spacecraft orbital requirements is listed in Table 1.

TABLE 1 – MAG CON ORBIT REQUIREMENTS

ITEM	REQUIREMENT
Number of Nanosat Min Max	 44 (2 per orbit) 104(4 per orbit)
Mission Orbits All Perigees Min Case Max Case Initial S/C separation # of Swarms (derived)	 3 Earth Radii (Re) 22 highly elliptical orbits, Apogees 10 to 52 Re (2 Re incr.) 26 highly elliptical orbits Apogees 10 to 60 Re (2 Re incr.) TBD 1
Mission Lifetime (years)	2
Inclination (Initial) Minimum Maximum	 7.5 deg 15 deg
Orbit Period Minimum Maximum	 0.97 days @ 10 Re orbit 10.4 days @ 60 Re orbit
Orbit Position Knowledge Science Communication	 +/- 20 km +/- 60 km
Eclipse (Duration) Time	 < 1 hr (greater if orbit plane in ecliptic)

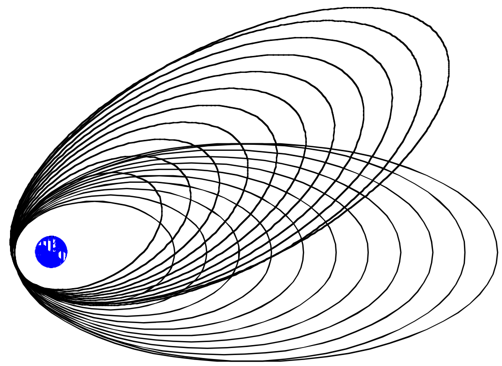
The ideal prototype mission would involve the simultaneous operation of multiple swarms of spacecraft. The baseline mission calls for approximately 100 spacecraft in one swarm. Such a constellation is proposed to survey the earth's magnetosphere during the Magnetospheric Constellation Mission. This mission would benefit from the breadth of variation in altitude and Mean Local Time in a relatively short period.

With the objective of making spacecraft sub-systems better, cheaper and faster, a prototype Magnetosheric Constellation mission scenario was defined in order to push GSFC's nanosat technology program efforts beyond current levels. Preliminary analysis was performed in order to obtain an understanding of the mission's potential orbital characteristics. The results are intended to spur the iterative design process of the spacecraft and the mission class. The following paragraphs summarize the mission design work to date. Table 2 enumerates the general requirements used in the analysis below. It should be noted that there are differences between the requirements in tables 1 & 2. This is because the analysis was performed prior to the updated requirements of table 1. It is believed that the analysis below is general enough to show the feasibility of the mission. The analysis will be redone in the near future to the requirements of table 1.

TABLE 2 – STUDY ORBIT REQUIREMENTS

ITEM	REQUIREMENT
Number of Nanosat	22 (2 per orbit)
Mission Orbits All Perigees Initial S/C separation # of Swarms (derived)	12 to 42 Re, 3 Re Incr. 11 Orbits 3 Earth Radii (Re) 1/2 orbit 2 (Arg of Perigee sep = 30 deg)
Mission Lifetime (years)	1
Inclination (Initial) Minimum Maximum	 1.0 deg 15 deg
Orbit Period Minimum Maximum	 29 days @ 12 Re orbit 6.25 days @ 40 Re orbit
S/C Mass	10 kg
S/C Size	30 cm dia x10 cm height
Orbital Position Knowledge	20 km
Eclipse Duration	475 min per orbit for 5 days

Unlike typical earth orbiting constellation designs, the focus of Nanosat is not planetary observation, but space science in-situ measurement. Thus, unlike other constellation concepts, there is an emphasis on widely varying orbit geometry. Such a configuration would provide science opportunities at locations about the earth with a considerable variation in observation parameters. There was expressed interest in very high apogee radii as well. This emphasis places a limit on launching many spacecraft in varying orbit planes, as the energy cost (e.g. propulsion) becomes prohibitive. Therefore the spacecraft in each swarm is designed to share one orbit plane at deployment.



Two Nanosat swarms, $\Delta\omega$=30
Figure 1

Within each swarm, by incrementing the apogee radius by 3 R_e, a total of 11 orbits are obtained ranging from 12 to 42 R_e . Another early concept was to deploy multiple spacecraft per orbit, in the hopes of obtaining an efficient distribution of science data. For simplification purposes, a distribution of two spacecraft per orbit was chosen. The spacecraft are assumed to be deployed in half orbit separations. Thus, a total of 22 spacecraft per swarm is obtained. Likewise, a second swarm was prepared, separated by 30° in argument of perigee (AOP) Figure 1. It is assumed, in this study that separate launch vehicles would deliver motherships for each swarm to a geosynchronous transfer orbit, 200 km x 6.6 R_e. An apogee kick motor would then raise the perigee of each mothership to 3.0 R_e while also maneuvering inclination and right ascension of the ascending node (RAAN). Each Nanosat would use on board propulsion to raise apogee into its respective orbit.

For the analyses below, the following analytic parameters were used. The force model used was the JGM-2 21 x 21 truncated earth gravity model. Solar Radiation was modeled using a Coefficient of Reflectivity of 2.0. Both variation of parameter and Runge-Kutta analytical propagators were used to predict long term effects of the earth and moon gravity model perturbations. The cross-sectional area of each spacecraft is 3.0×10^{-8} km^2.

As Nanosat matures in concept, undoubtedly requirements will change. However an effort was made in this mission design to quantify the environmental factors pertinent to spacecraft design. The preliminary studies presented herein are in response to concerns of the project spacecraft subsystem designers.

For such an ambitiously small spacecraft design, challenging constraints on power are imposed. Consequently, the first analysis performed was a survey of eclipse duration for the worst case orbits. Consideration of design criteria evolved from that analysis to the launch opportunities afforded by minimizing eclipse conditions. In response to the need for sizing the command, communication and control subsystems, an investigation into ground contact opportunities was initiated. As a follow on to these results a quick estimate of orbit determination accuracy was performed.

ECLIPSE DURATION

In studying the eclipse duration maximums of Nanosat spacecraft, it is assumed that the worst case conditions exist for orbits with small inclinations and the longest orbital periods, e.g. the 40 R_e apogee case. Nonetheless it is helpful to examine the eclipse trends of other orbits in relation to various orbital parameters.

This study surveyed maximum eclipse duration for a variety of parameter values. Inclinations of 1.0°, 7.5°, and 15°; apogee radii of 12, 26, and 40 R_e; ascending nodes of 0, 90, 180, and 270°; and epochs of March 21, June 21, September 21, and December 21, 2008 at 00:00 GMT were all observed. For each combination of values, a one-year propagation was conducted. The maximum eclipse duration for these parameters are presented in Figures 2 to 10. As expected, the 40 R_e apogee radii for all three inclinations are the worst cases where the maximum eclipse duration exceeds 500 minutes (Figures 8, 9 & 10). In most cases maximum eclipse duration occurs at values of either 0° or 180°. For these same high apogee orbits values of 90° or 270° reduce the maximum eclipse by about one half to under 300 minutes. When the apogee radius is reduced to 26 R_e for a similar set of initial conditions, the maximum eclipses are upper bounded by 350 minutes with RAANs of 0° and 180°, again, causing the largest eclipses. The cases for RAANs of 90° and 270° show maximum eclipses less than or equal to 160 minutes. For an apogee radius of 12 R_e, the values for bounding the maximum eclipse duration are 150 minutes and 90 minutes for RAAN pairs of (0° and 180°) and (90° and 270°) respectively. From these cases it is concluded that choosing the RAAN judiciously will permit the spacecraft design to incorporate the effects of a much lower maximum eclipse.

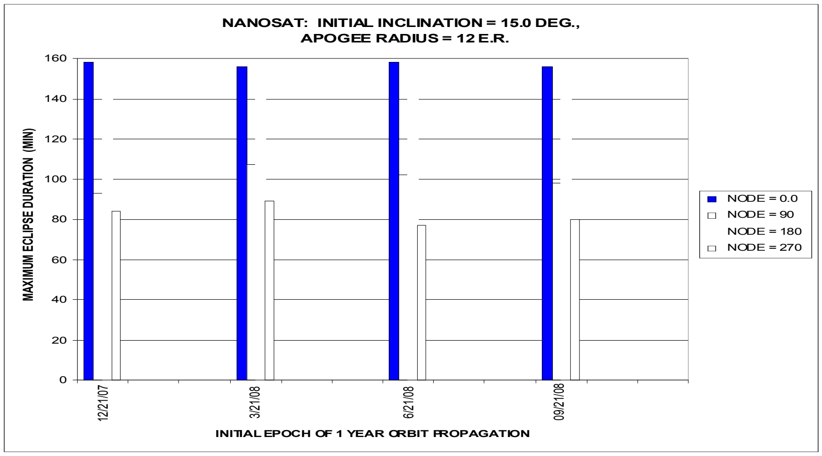


Figure 2

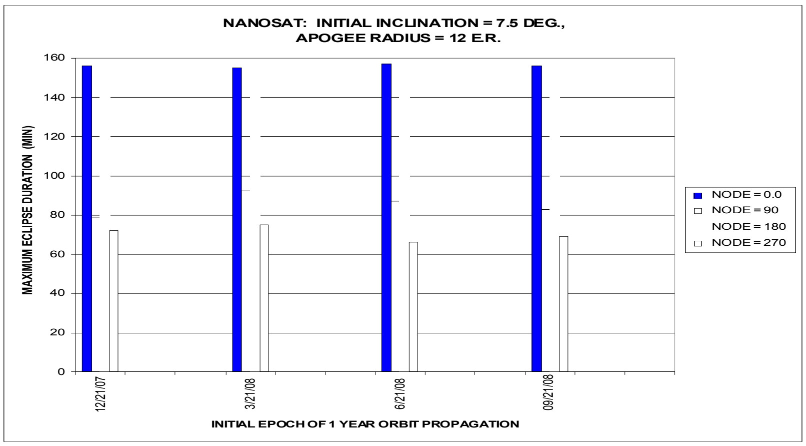


Figure 3

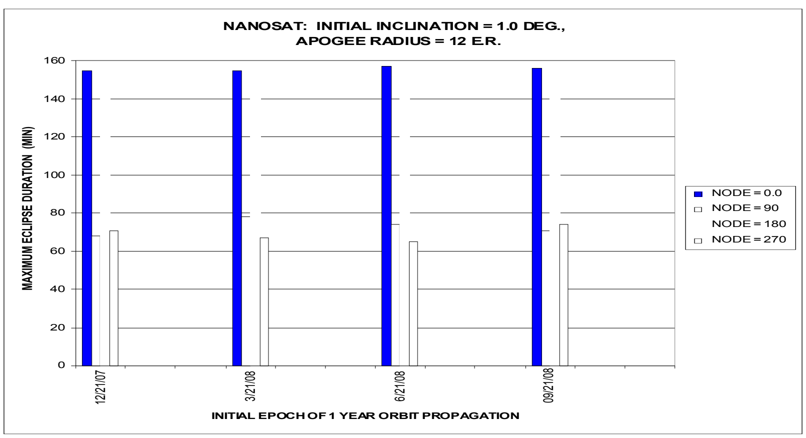


Figure 4

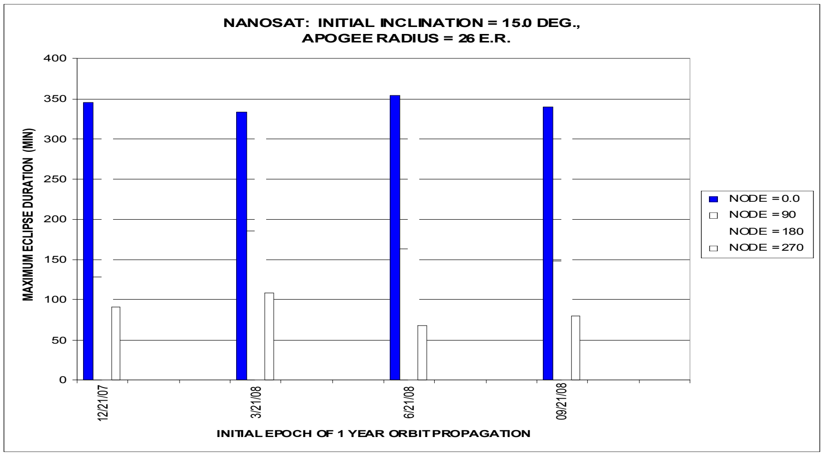


Figure 5

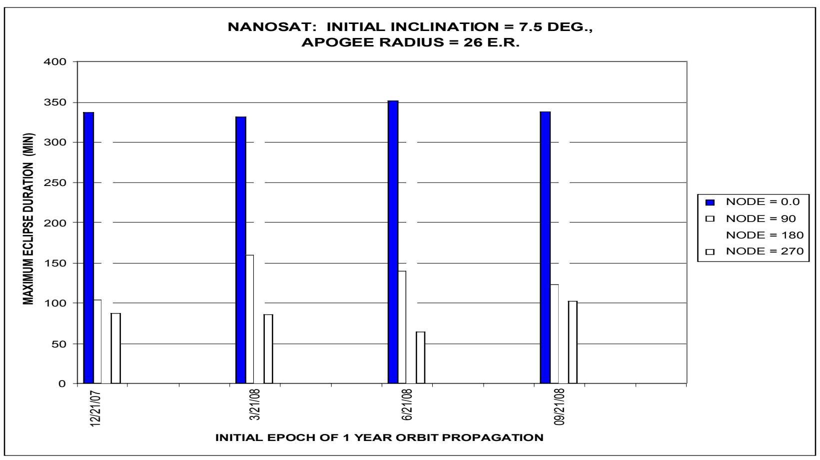


Figure 6

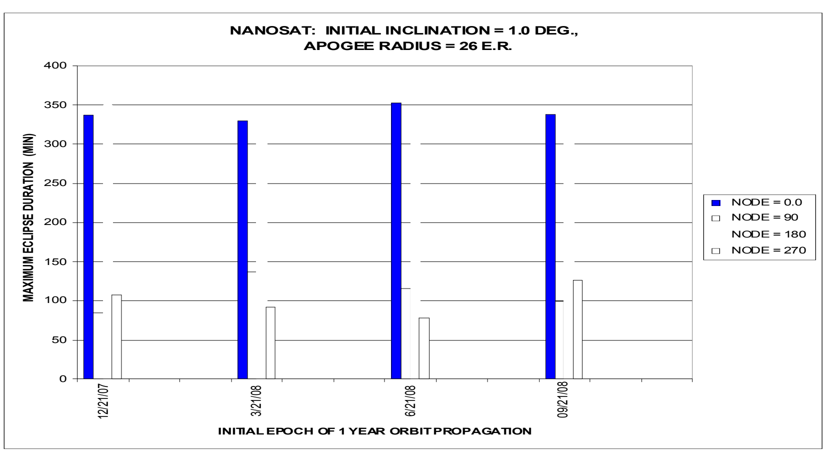


Figure 7

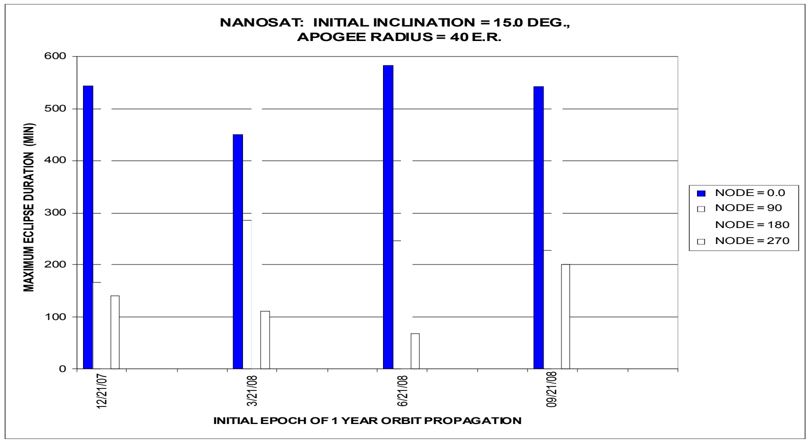


Figure 8

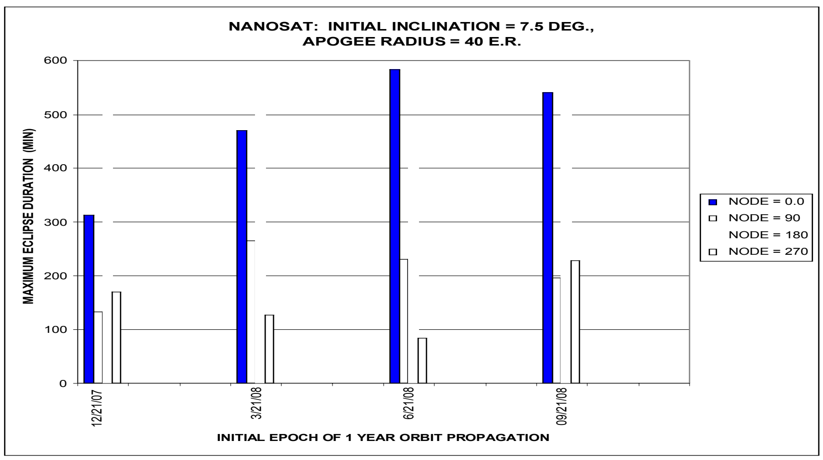


Figure 9

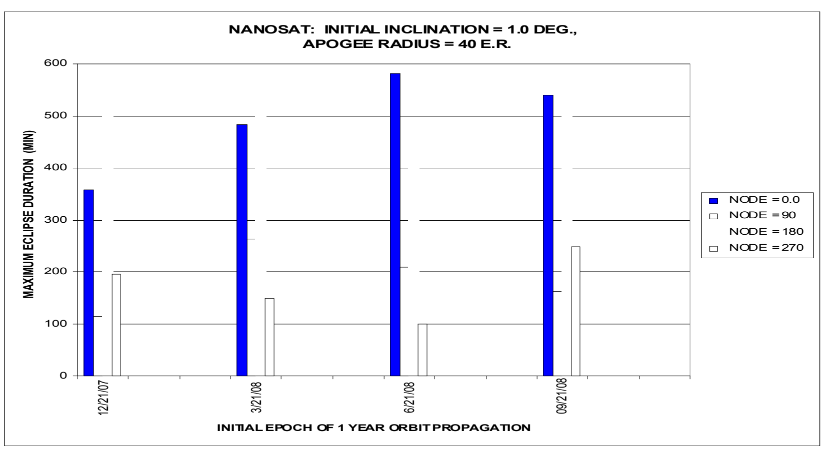


Figure 10

For spacecraft thermal control considerations, shadow conditions in excess of 400 minutes were deemed too conservative by subsystem engineers. Early power estimates established an eclipse restriction of 475 min per orbit for 5 days. These results show that these constraints can be managed, with compromise in initial orbital element selection. The RAAN cases for 90° and 270° are the most favorable and satisfy these constraints.

While the primary focus of this study was identifying eclipse trends, it is worthwhile to note the spacecraft state at the end of one year. Most notable is the orbit perturbation effect of the lunar gravity. The chief consequence of this effect is a large change in inclination for the high apogee orbits. Figure 11 shows the inclination history over the course of one year in the absence of correction maneuvers. Depending on the epoch, the inclination can increase as much as 30° over the course of a year in this configuration. This stands to reason as the 40 R_e orbit semi-major axis is a significant percentage of the Earth-Moon distance (36% mean lunar distance), and therefore highly susceptible to lunar gravity orbit perturbations.

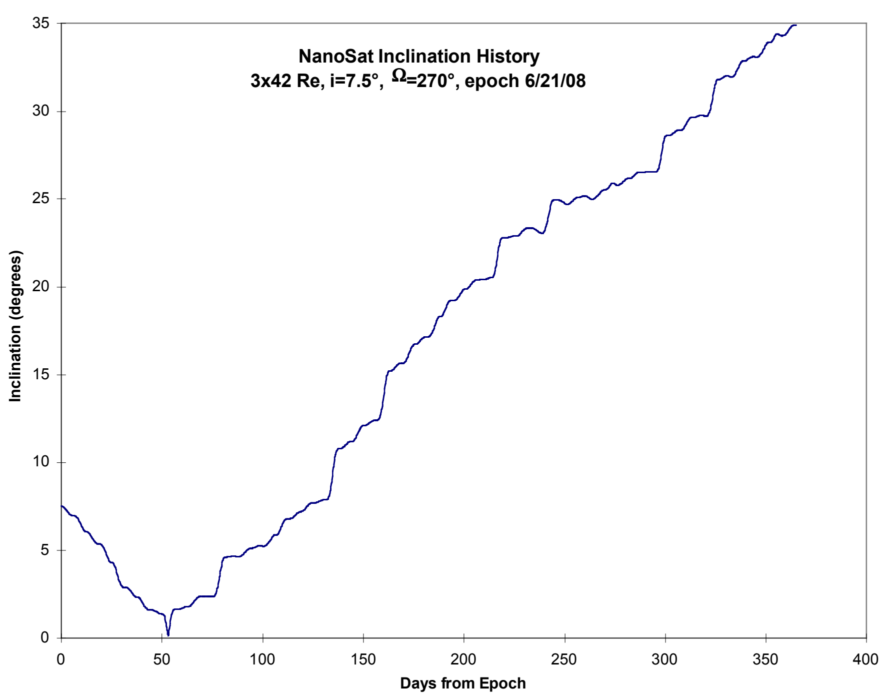


Figure 11

LAUNCH OPPORTUNITIES

In consideration of the eclipse study results, an examination of launch opportunities over the course of one day was computed for inclination of 7.5°, apogee radius of 40 R_e , and an epoch of June 21, 2008. As was shown in previous work (Figures 2 to 10) the summer solstice cases tended to have lower overall maximum eclipse values. Although the 15 degree inclination orbits yielded slightly better eclipse results over the course of one year, the 7.5 degree case was chosen in order to retain favorable field of view margin for payload instruments.

As before, the parameter of note for this one day study is RAAN. Due to the rotation of the earth, RAAN at launch varies by 15° per hour. Launch opportunities at 2 hour intervals were examined. In each case a 1 year propagation was performed to evaluate the maximum eclipse duration. For comparative purposes, another swarm was modeled by introducing a 30 degree difference in AOP, and generating another data set. Results are presented in Figure 12. The magnitude of the maximum eclipses followed the trend seen earlier with the initial RAANs near 90° and 270°: yielding smaller maximum eclipse durations than those with RAANs near 0° and 180°.

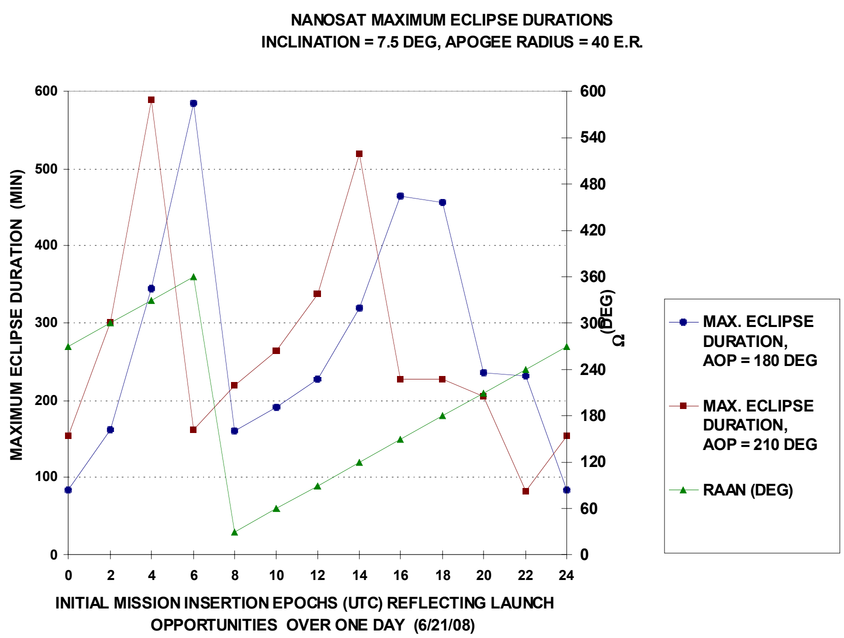


Figure 12

One last set of calculations for the Nanosat eclipses showed the distribution of eclipse durations over one year. For an initial _ of 270°, inclination of 7.5°, and an apogee radius of 40 R_e, four different one year propagations were made each beginning with a different epoch, corresponding to equinoxes and solstices. Results are shown in Figures 13-16. The choice of RAAN at 270° tends to reduce the maximum eclipse for these conditions. The placement of the first local maximum for the eclipse seems to coincide approximately with the next equinox. Later local maximums do not necessarily hold to an equinox condition due to the orbit precession.

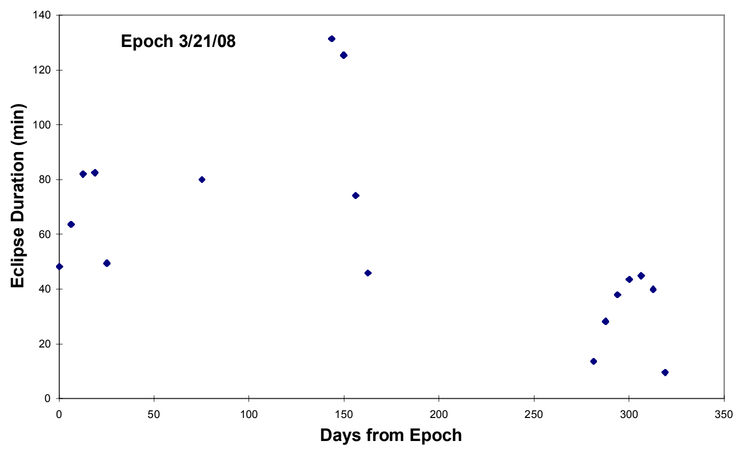


Figure 13

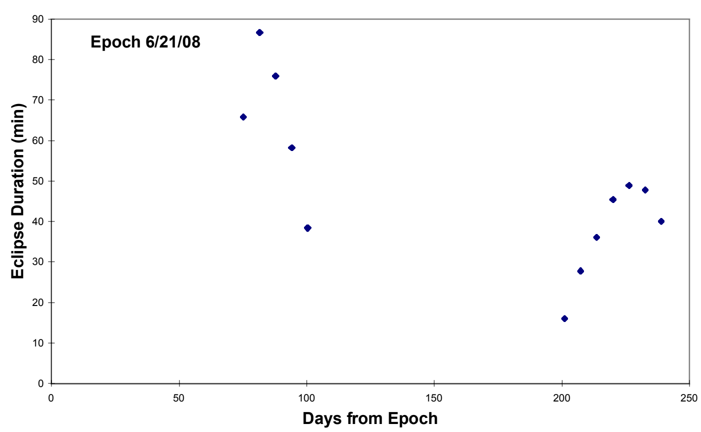


Figure 14

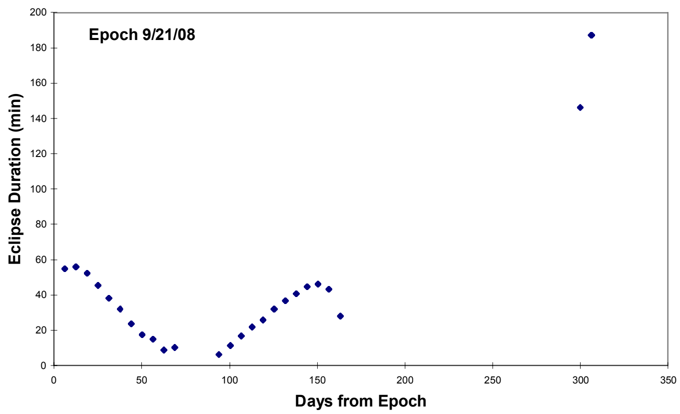


Figure 15

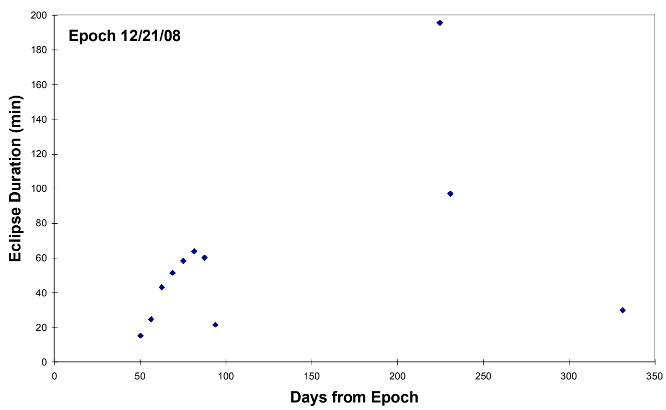


Figure 16

CONTACT SURVEY

Using the eclipse and launch window analyses results, the prototypical set of orbits was selected for analyzing the ground station coverage for the entire Nanosat constellation. The two swarms prepared for the study were offset by 30° in AOP. Each swarm had RAAN of 270°, inclination of 7.5°, epoch of June 21, 2008 at 00:00:00 GMT. One swarm had an AOP of 180°, while the second swarm had an AOP of 210°. Each swarm consisted of 11 orbits. The two resident spacecraft in the 3x12 R_e were separated initially by 180° in mean anomaly. For each successive orbit, the next two spacecraft initial mean anomalies were incremented by 15°. These initial conditions were chosen in an attempt to pseudo-randomly distribute the swarm spacecraft. In total, 44 spacecraft were examined.

For providing typical global coverage, three ground station locations were selected. Antennas were placed at Goldstone, Madrid, and Canberra, each assuming a 5 degree elevation mask. Use of these facilities by Nanosat is not implied by their inclusion in this analysis. A slant range limit of 5 R_e was used, to take into account the power limitations of the on board antenna.

Over the course of one year 11,500 passes were simulated over the three stations. In this period the busiest ground station coverage event was simultaneous contact with 12 spacecraft, which was highly anomalous. Figure 17 shows a 200 day history of simultaneous contacts sorted by station. This data was used by ground station schedulers to estimate the required data rates and storage specifications for Nanosat spacecraft. For the periods of high number of spacecraft contact, the lower orbits may be considered lower priority for link scheduling, as a download opportunity may be delayed for a relatively shorter period of time (3x12 period = 29 hours). In contrast, the high orbit periods consist of several days, (3x42 period = 6.25 days) offering long wait times in the event a link is not established.

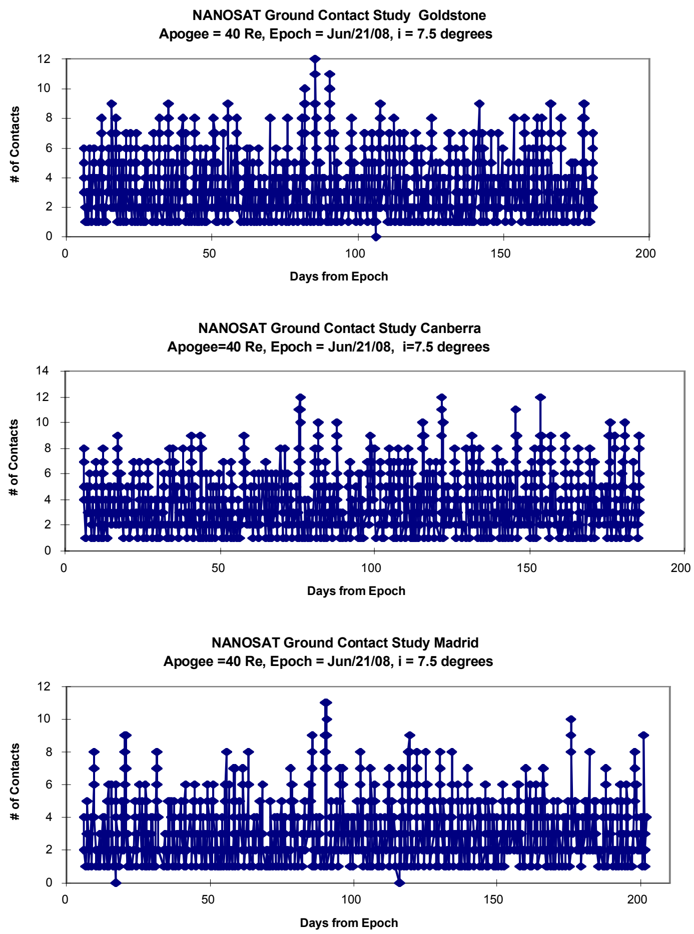


Figure 17

ORBIT DETERMINATION/ERROR ANALYSIS

Setting the upper bound in error scenarios, the largest orbit is selected for the purpose of error analysis in orbit determination. The pass statistics generated in the last section were scheduled for the following scenario. A 40 R_e apogee radius Nanosat contacts one ground station per pass and is propagated for the remainder of its period until contact is re-established. For the purposes of this analysis the assumption is that the time between passes equals the orbit period minus the contact time. For one ground station, this is not necessarily true for all cases. The assumption depends on the coverage of the other two ground stations. Contact times will also vary in length, therefore a survey of possible contact times between one ground station and one spacecraft using Doppler range-rate measurements was conducted. Since the period of this orbit is much greater than any of the contact passes, there is no significant variance in orbit determination error. A pass duration of 120 minutes yields a total position error of 19.2 km, total velocity error of .388 meters/sec. A pass duration of 300 minutes yields a total position error of 18.9 km, total velocity error of .380 meters/sec. The larger duration passes are available when slant range constraints are loosened, however there is no appreciable advantage using this technique since the predominant source of error in these ranges is the contribution of tropospheric refraction of the measurement.

CONCLUSIONS

From the survey of orbital parameters conducted in this analysis, a workable set of design options is obtained. A feasible orbit design, though preliminary, may be selected given the constraints of the mission. In order to minimize the maximum eclipse duration for the longest orbit period, right ascension of ascending node of 270° is recommended. By selecting a launch epoch near June 21, 2008 and scrutinizing Figure 10 for best launch opportunities over the course of one day, one year maximum shadow conditions can be designed well within spacecraft requirements. A one year mission is feasible with the current requirements, if large inclinations at high orbits can be accepted. A ground system must be designed to accommodate several simultaneous contacts over the course of one year, where the maximum observed can be 12 spacecraft. Orbit determination error can nominally provide a state within 20 km of accuracy for the largest orbits.

The results reported in this survey were generated using Analytical Graphics Inc.'s Satellite Tool Kit software, version 4.0. The Goddard Space Flight Center's Goddard Mission Analysis System (GMAS) and Orbit Determination Error Analysis System (ODEAS) were also used.

NEW FINDING FOR 60 Re ORBIT

This addendum to the Nanosat Constellation Orbit Analysis addresses the proposed expansion of mission orbit apogees to 60 Earth Radius (R_e). The above analysis examined a range of Earthcentered orbits from 12 to 42 R_e, with perigees of 3 R_e. The most dramatic effects of the proposed 60 R_e apogee on shadow duration, inclination history, and mission lifetime are included. This new apogee goal is shown to be largely unfeasible for the baseline mission lifetime.

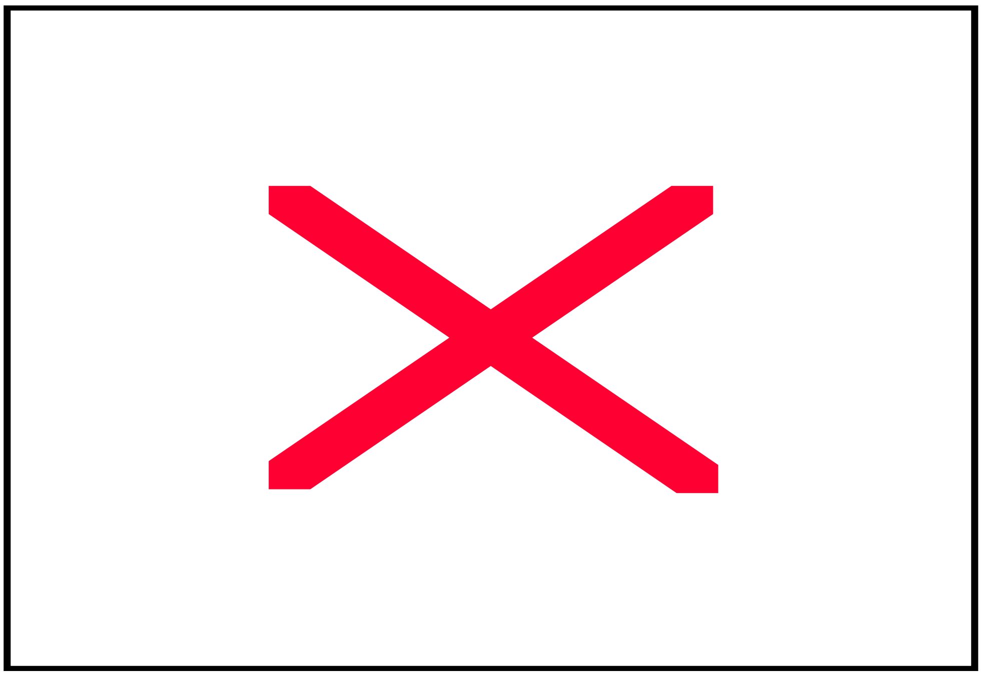
For the case of a 3x60 Re Nanosat orbit, inclined 15 deg to the equator, several issues are worthy of note:

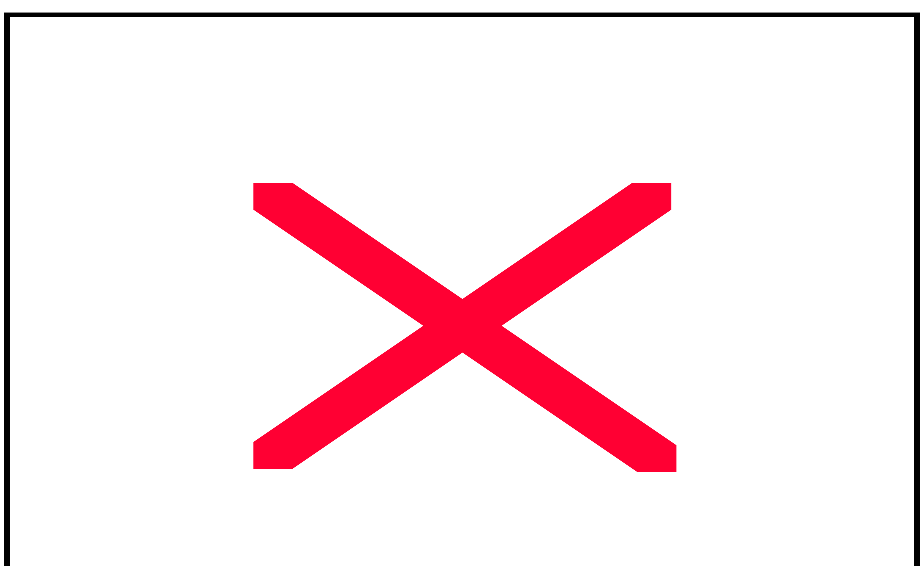
1. At 60 R_e, the apogee closely approaches the average Earth-Moon distance, and may represent under the proper circumstances, a lunar swing-by.
2. A first order approximation of the eclipse duration at 60 R_e at worse case is very high, 11.25 hours.
3. The inclination of such an orbit is likely to evolve outside of operational goals within half of the desired mission lifetime.

The Earth-Moon average distance is 384,400 km over the calendar year. An apogee of 60 R_e represents a distance of (6,378.14 x 60) = 382,688.4 km. Taking into consideration that the Moon's radius is an average of 1,737 km, a perfectly aligned orbit would represent a lunar impact condition! This assumes the line of apsides lies in the Moon's orbital plane, and remains there over the course of the mission. Even if the 60 R_e Nanosat orbit was placed away from the Earth-Moon line, the different periods of the moon and 60 R_e Nanosat would ensure a lunar impact within 280 days, the approximate synodic period of the Moon and the 60 R_e Nanosat. The peril of implementing such a near lunar trajectory is the possibility of lunar impact or the very real possibility of Earth escape via a lunar "swing-by " trajectory. The lunar sphere of influence extends out 66,200 km from the Moon. The possible effects of a swing-by within this sphere include drastic inclination change to polar orbit, velocity gain that raises the orbit to an Earth escape trajectory, impact, or periapsis lowering or raising to unacceptable levels. Clearly, these outcomes jeopardize the mission.

Repeating the assumptions of the line of apsides above, a first order approximation of eclipse duration is (Earth diameter / apogee velocity) = (12756.3 / .31496) = 11.25 hours. While it is more likely that apogee will be well off the ecliptic plane and the umbral cone is less that the earth's diameter at the lunar distance, this is nonetheless a good first approximation for worst case scenario consideration.

As reported in the analysis in the previous section, the inclination limit outlined for the Nanosat mission is 15 degrees. This analysis used 7.5 degrees as an initial condition. In previous analyses, the 3x42 R_e orbit was shown to change inclination due to lunar perturbation by 20 degrees in one year. A similar analysis shows that the 3x60 R_e orbit inclination changes by 30 degrees after only 180 days. The dominant perturbation force is the lunar mass. This effect is exaggerated in this case because of the proximity of the Moon. Figure 1 shows the inclination history of a 3x60 R_e Nanosat initially inclined at 7.5 degrees. Figure 2 illustrates the effect of the inclination on the distance the Nanosat travels out of the equatorial plane.





The new apogee goal of 60 R_e, largely because of undesirable lunar effects, is shown to be unfeasible for the baseline mission lifetime. The inclination and eclipse duration rise above acceptable levels, but most importantly, the proximity to the Moon raises some serious concerns about the viability of the mission lifetime. For these reasons, the maximum apogee recommended would extend to the periphery of the lunar sphere of influence. This maximum is equivalent to an apogee of 50 R_e. Any orbit apogee design that is larger than this value will have to incorporate a scheme to avoid the obstacles presented herein. Possible measures include raising the design perigee to increase the orbit period, and thus the synodic period, reducing the lifetime goals to incorporate a limited operational condition, and selective placement of the line of apsides to avoid maximum shadow.

Future mission tradeoff studies will have to address the final Magnetospheric Constellation maximum apogee with the above in mind.

ORBIT RADIATION ISSUES

In order to evaluate the potential danger of a spacecraft in a 1.5 ER perigee orbit, research was performed on the location and intensities of the radiation belts, as well as other radiation hazards. In addition, a meeting was held at GSFC with E. G. Stassinopoulos, Pete Panetta, and other subsystem leads, to discuss the subject.

Dr. Stassinopoulos discussed the location of the radiation belts, including the high energy protons and electrons at 1.5 Re. He stated that at 1200 km, the atmosphere has drag, depending on solar activity. This effect, plus the influence of the moon, sun, and other planets, will cause apogee and perigee shift, as well as inclination changes.

Outside of the radiation belts, there are still cosmic ray protons (H atoms), and solar events, such as flares and corona mass ejection's, both of which eject large number of energetic protons, such as 10.0E+10 protons with energies > 30 MeV. Solar cycles average 11 years, with a variation of +/-2 yr. Solar maximums are roughly 7 years, and are in the middle of the cycle, with solar minimums averaging 4 years. Solar cycle 22 had 8 mass ejection's in one year alone. A new solar cycle has begun, and the next year to be active is expected to be 1999. Years 2003-2004 will be in the middle of the next solar maximum, but even during maximum there can be very quiet years, and very active years.

In terms of design criteria, some people have used 100 krads total dose as a design criteria, but Stassinopoulos's opinion was that there are very few parts that by themselves are capable of this, and that even if they were, this is not enough protection against large solar events. They recommend determining the sensitive components, and then locating them so as to use existing structure, or the addition of tantalum as need, to shield those parts. For electrons he said that on the CRES mission, over 14 months, they saw 800 krad on the outside of the spacecraft, 38 krad on the outermost board, and on the inner most board 7 krad.

They recommended that an analysis be performed when the orbits have been finalized, as well as strongly advising that the perigee be raised to 3 Re. They could at that time perform an analysis to determine the worse case orbits, and then provide the trapped electron and proton numbers, as well as the solar proton numbers.

LAUNCH VEHICLE

The launch vehicle assumed for the MAG Constellation mission is the Delta II 7925 Launch Vehicle with a 9.5 ft fairing. A pictorial of this rocket is shown in figure 4.This vehicle is a 3 stage vehicle. The 1st stage has a liquid engine with 9 attached solid propellant graphite epoxy motors. The 2nd stage also has a liquid rocket engine. The 3rd stage carries the MAG CONSTELLATION Payload (Deployer Ship and the complement of Nanosats) which is attached to the 3rd stage via a clamp band separation system. The launch vehicle will separate the Deployership placing it and it's Nanosats into a 1000 km x 20 ER orbit.

MISSION OPERATIONS

Magnetosphere Constellation Operations Concept

The operation of a constellation of identical nanosatellites requires different concepts than for single spacecraft missions. The nanosatellites may be constrained in the amount of functions they can perform onboard by the lack of processing power. The large number of spacecraft requires automation on the ground in order to keep the staffing to a reasonable level. The large number of identical spacecraft also offers an opportunity for changes in risk management. The loss of a few spacecraft over the mission lifetime is acceptable for this constellation.

Figure 1 shows the ground data system for the Magnetosphere constellation. The ops center and the science data systems are shown as separate standalone systems in this figure. The actual implementation may combine these functions, or perform these functions in centers and systems that support other missions.

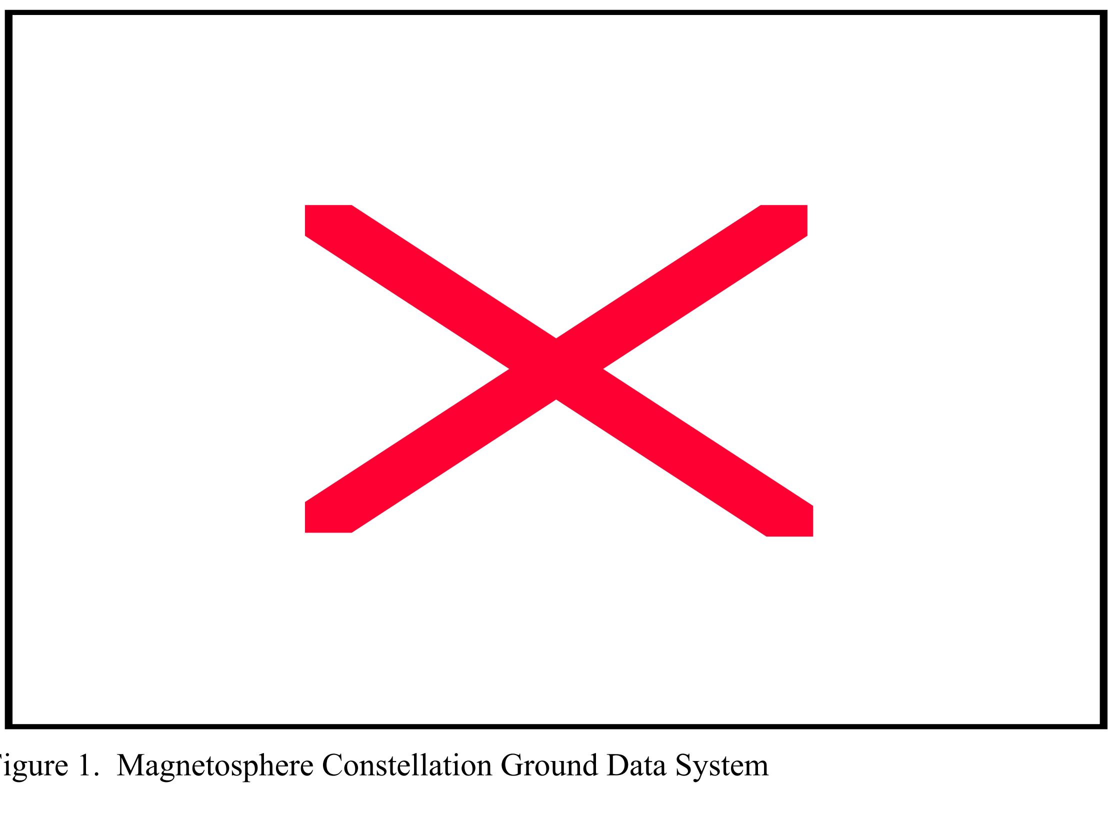


Figure 1. Magnetosphere Constellation Ground Data System

Concepts for Routine Operations

Routine operations commence after the nanosatellites have reached their operational orbit. The routine operations are only interrupted by anomalies or infrequent events such as eclipse.

Space/Ground communications. There are several options for space/ground communications services - dedicated stations, commercial networks, NASA stations, large antennas, smaller antennas, and so on. This ops concept assumes the use of a commercial network of 11 meter antennas. The commercial network would have a number of these antennas distributed around the world. Communications would only occur around perigee. At some points in the mission, as many as two dozen spacecraft might be near perigee at the same time. The multiple stations would be able to handle some of these spacecraft. Spacecraft with smaller apogees could skip a contact, and dump their data at the next perigee. Some data can be lost - the overall data completeness requirement will be about ~95%. Loss of some data during these infrequent "traffic jams" would be acceptable.

The ground stations will provide telemetry, commanding, and tracking services. The real time housekeeping data will be extracted from the downlink data and sent to the operations center in real time. Playback science and housekeeping data and tracking data will be sent to the operations center after the contact, sharing the bandwidth with other station users.

Planning and Scheduling. The instruments are expected to have several modes - a normal mode, a high rate mode, and a mode where the instrument does not take data. The science users will define the sequence of modes for the instruments. The sequence may be changed during the mission. The science users will provide the new sequence to the mission operations team, which will convert them to command loads and schedule them to be uplinked within some time window.

The scheduling of the ground stations depends on how the ground station services are provided. One option would be to let the ground station provider schedule the contacts within Magnetosphere Constellation provided guidelines for data completeness and frequency of tracking data collection. The ground station provider would have the flexibility to adjust the contacts within these guidelines, which potentially could result in lower costs to the project. The ground station provider would provide the schedule to the operations team, to support the identification of uplink opportunities and to support data accounting.

Commanding. The spacecraft will be commanded every contact to initiate the downlink of data. Since the ground station schedule is subject to change, particularly for the spacecraft with distant apogees, the downlink could not be initiated by the spacecraft.

The instrument scripts may be updated once every month or so. The instrument script may be as simple as timed tagged commands or could be a more sophisticated set of instructions for the onboard system to interpret.

Software loads may be performed if problems are identified after launch that can be addressed by the flight software.

The data system will automatically uplink the new loads or instrument scripts to the constellation. The loads will have a time window within which each spacecraft must be updated. The system will automatically send and verify the load during regularly scheduled contacts. The operators will be alerted of any spacecraft that is not updated when the window closes.

Housekeeping Data Processing. The data system will automatically process the housekeeping data and identify limits violations and misconfigurations. The system will be capable of processing data from multiple spacecraft simultaneously. The data system will also process status information from the ground station and identify any missing or poor quality data. Problems will be ranked according to severity. Significant problems will result in an immediate alert to the operations team. If the operations team is not present, the system will use a beeper to notify them.

The data system will process all of the housekeeping data and produce standard analysis reports. These reports will compare telemetry parameters across time and across spacecraft. Spacecraft engineers will be able to generate custom reports as well.

Science Data Processing. The science data is automatically level zero processed when it is received from the ground station. This processing removes overlaps in the data and identified missing data. Operators are alerted in the event that a significant amount of data is missing. The system will have data accounting tools to assist the operators in visualizing the data loss and its distribution over the constellation. The operators will work with the ground station network to attempt to recover lost data that exceeds requirements. They may adjust the scheduling priorities if some part of the constellation is losing more data than other parts.

The data will be automatically processed into standard products and archived in the science center. It will be distributed to science users electronically or, for large amounts of data, on physical media.

The science data systems will have tools to assist the users in subsetting the data from the entire constellation in meaningful ways. It will have visualization tools for viewing constellation data.

<u>Orbit Determination.</u> There are several options for orbit determination under consideration. They are more fully discussed in the orbit determination section. The concept briefly discussed here assumes that each spacecraft has an ultra-stable oscillator and the orbit is determined on the ground using one-way doppler.

The operations center receives the tracking data from the ground station and automatically processes it to determine the orbit. It updates the acquisition data and provides it to the ground station network. The orbit data is also provided to the science data processing system.

The automated system will alert an operator if tracking data is not received from a spacecraft within a certain period of time. It will also alert the operators to potential quality problems with the data.

<u>Operations Staffing.</u> The mission operations staff (not including the science ops staff) for the magnetosphere constellation can be relatively small. The spacecraft are performing survey missions and have limited reconfiguration capability. The ground systems are automated, performing all routine functions. The ops team will primarily handle exceptions that have been flagged by the data system. The ops teams primary concern will be for common problems that have the potential to affect all of the members of the constellation. Random failures of a few spacecraft over the mission lifetime will be acceptable.

The nominal ops staff will be about 6 people. It will include a couple of spacecraft engineers, someone who will work with the science users, a data accounting person, and an orbit expert. The ops staff will work 40 hours per week, and at least one member will be on call for automated alerts during off-hours.

SPECIAL OPERATIONS

At launch, the operations team will be augmented with personnel from the spacecraft development and test functions. The ground data system may include more ground stations or larger ground stations, to checkout and track the spacecraft after deployment. The ground system will also have to operate the deployer ship during this phase of the mission. The deployment operations may be adjusted (for example, to change the number of spacecraft deployed at a time, or change the frequency of deployment), based on the experience with the first few deployments.

The nanosatellites may infrequently experience long eclipses. The impact of these long eclipses on the power and thermal systems could be significant, depending on the actual design. The operations team may take special actions to configure the nanosatellites for eclipse, and perform special evaluations of the spacecraft after the eclipse, to identify any change in performance.

In the event of an anomaly, the ops team may request the assistance of the spacecraft developers to identify the cause of a problem and to develop corrective action or work arounds. The spacecraft developer will be responsible for maintaining the flight software throughout the mission. To assist in the anomaly resolution, the ops team may use larger ground antennas to communicate with the spacecraft over longer periods of time, rather than just at perigee.

MISSION INSTRUMENTS

SEC Magnetospheric Science Objectives

One of the great achievements of space science has been the determination that the "magnetospheric substorm" involves a complex chain of plasma processes beginning with the transfer of energy from the solar wind to the magnetosphere at the dayside magnetosphere and its subsequent, explosive dissipation in the tail region. This sudden release of large amounts of energy, ~10^{10} - 10^{11} W for intervals of several hours, in the magnetotail has many effects on the Earth's environment ranging from the generation of beautiful auroral displays in the high latitude ionosphere to the acceleration of large fluxes of energetic particles which can have deleterious, or even fatal consequences for satellites. More than any other phenomenon except solar flares, the magnetospheric substorm has demonstrated the remarkable ability of cosmic plasmas to concentrate large quantities of energy in strong magnetic fields and then quickly convert it to charged particle, electrical and hydromagnetic wave energy. The process responsible for these episodes of explosive energy release is “magnetic reconnection.” The understanding of the reconnection process through which solar wind energy is acquired at the dayside magnetopause and later released rapidly in the tail is a cornerstone science objective of NASA's Sun-Earth-Connection (SEC) research programs.

Technology Driven Mission Concepts

The ability to fabricate, deploy and operate large numbers of nanosatellites flying in pre-determined constellations has been identified by NASA Headquarters and the SEC community as essential to obtaining definitive observations of magnetic reconnection and other dynamic plasma processes. A Specific future mission requiring this capability is Magnetospheric Constellation (MagCon) Solar Terrestrial Probe (STP) Mission.

The highest priority instrumentation to be carried by MagCon are magnetometers, plasma analyzers and energetic particle detectors. Miniaturized versions of such instruments have been under development for some time at a number of academic, commercial and government laboratories.

The successful implementation of such instruments levies certain non-interference requirements upon the spacecraft and the instruments themselves. For example, the magnetometer measures the magnetic fields due to all sources including those arising from the spacecraft and the instruments it carries. Similarly, the plasma instrument observations can be adversely affected by the build-up of electrostatic charge in the vicinity of its apertures and the energetic particle analyzer is sensitive to any onboard radioactive materials. All three instruments are also susceptible to conducted or radiated electromagnetic interference originating with the spacecraft subsystems or the other instruments. Many of these accommodation issues become more challenging as the size of the satellite decreases and the proximity of the instrument sensors to the spacecraft and instrument electronics grows.

MagCon will require the efficient fabrication, deployment and operation of multiple small spacecraft in pre-determined constellations. Techniques utilizing particles and fields measurements enable structures (e.g., current sheets) and fluctuations (e.g., plasma waves and shocks) to be characterized far more fully than is possible with a single spacecraft.

Vector Magnetometer Instrument Requirements

Highly accurate and sensitive measurements of the orientation and intensity of the ambient magnetic field is critical to the success of the Magnetospheric Constellation mission. A knowledge of the magnetic field is fundamental for the understanding of the structure and dynamics of the magnetosphere. Thin current sheets, such as the dayside magnetopause and the cross-tail current layer on the nightside, are the primary sites where solar wind energy is first converted to magnetic energy and then later dissipated in the tail via reconnection. In addition, low frequency (< 1 Hz) MHD waves and steepened waves sometimes resembling shock fronts must be accurately detected and analyzed due to their important contributions to the heating and acceleration of charged particle populations in the near-magnetotail.

The magnetospheric magnetic field in the outer magnetosphere (i.e., beyond geosynchronous orbit) ranges from a few 100 nT closer to the Earth down to order 0.1 nT in the vicinity of the cross-tail current layer. However, for ease of ground-checkout and handling and the desirability of being able to evaluate spacecraft sensor performance and spin axis attitude in the high fields encountered over a range of possible perigees, the Mag Constellation vector magnetometer should have sufficient dynamic range to measure magnetic fields as great as those encountered at the Earth's surface ~32,000 to 64,000 nT. An accuracy of 0.1 nT and a sensitivity to perturbations as small as 0.01 nT in an ambient magnetic field of 100 nT is necessary in order to fully characterize the magnetic field structure and waves present in the outer magnetosphere.

The triaxial fluxgate magnetometer technique which has been used in the exploration of the heliosphere and the characterization of planetary magnetic fields (e.g., Mariner 10, Voyager, Pioneer Venus Orbiter, Mars Global Surveyor, Galileo, Lunar Prospector, etc.) is well suited to measurement requirements of the Mag Con mission concepts. Research grade, ultra-low mass/power, miniaturized ring core sensors with dimensions of only a few cm on a side have been developed and tested to the extent possible in the laboratory. However, they lack extensive space flight heritage and validation in space. Total mass and power consumption, including all analogue and digital electronics for such a fluxgate magnetometer with 2 sets of triaxial sensors are under 1 kg (excluding boom) and 1 W, respectively. Furthermore, the availability of radiation hardened, fast, low power 16 bit analogue to digital converters should allow sufficient digital resolution with, at most, only 2 ranges which simplifies the in-flight calibration and ground-processing. Finally, a triggered burst memory should be included within the vector magnetometer (or the spacecraft electronics) to allow for autonomous decision making and the triggering of the high time resolution burst mode during dynamic events such as encounters with magnetospheric boundaries. These high rate measurements would be stored for later play-back during normally scheduled telemetry passes.

Plasma Analyzer Instrument Requirements

(*This instrument has not been included in mission studies up to this time*)

The Earth's magnetosphere is the contemplated target for the Mag Con nanosatellite mission, and it is crucial that this mission provide sensitive measurements of the highly variable magnetosphere plasma environment. The nature of the magnetospheric plasma varies considerably from one region to the next. Measurements inside of ~5 earth radii will be extremely difficult at low latitudes because of contamination by the penetrating radiation from the radiation belts. Beyond that distance the regions encountered will be the plasma sheet, the plasmasphere, the tail lobes, and the magnetosheath. Each region has a characteristic range of densities (n) and temperatures (T) approximately as follows:

Region	n (cm^{-3})	T_e (K)	T_p(K)
plasma sheet	0.1 - 3	$1 \times 10^{+6}$ - $5 \times 10^{+7}$	$5 \times 10^{+6}$ - 10^{+8}
plasmasphere	0.1 - 10^{+3}	$5 \times 10^{+3}$ - 10^{+5}	$5 \times 10^{+3}$ - 10^{+5}
lobe	.01 - 10	10^{+4} - 10^{+6}	10^{+4} - $5 \times 10^{+6}$
magnetosheath	1 - 200	$5 \times 10^{+5}$ - $5 \times 10^{+6}$	10^{+6} - $2 \times 10^{+7}$

In order to characterize the plasma in these regions measurements of both ions and electrons will be necessary. The measurements should extend from a few eV (the lower the better) up to about 30 keV for both species. The desired field of view for the plasma measurement is 4 pi steradians. These measurements are best made on a rapidly spinning spacecraft with its spin axis normal to the plane of the ecliptic. Spacecraft charging and its effect on the low energy particle measurements will be major issues.

Recent advances in the development of low resource budget plasma instrumentation suggests that a minimum measurement can be made with an experiment having a mass of 1 kg, a power consumption of 1 Watts, packaged in a box 8 - 10 cm on a side. The required telemetry for the plasma analyzer ranges from as low as 100 bps to 1-2 kbps depending upon the mission science objectives, the amount of onboard data processing and compression, and the availability of a burst memory.

The above types of instrumentation will not be able to distinguish between the different ionic components of the magnetospheric plasma. Nevertheless, some of the most fundamental questions about the magnetosphere, including questions about the relative strengths of the solar wind and ionospheric sources in the inner magnetosphere, require plasma composition measurements capable of separating at least H^+, He^+, He^{++}, and O^+ for their resolution. We are not aware of presently available plasma composition instrumentation for this measurement that can come close to fitting into the above noted mass, power, and size envelope. However, the development of such plasma composition instrumentation in conjunction with the Mag Con nanosatellite mission would be highly desirable.

Energetic Particle Detector Requirements

The Mag Constellation mission will fly a highly miniaturized yet capable energetic particle detector system. Furthermore, it will demonstrate that this sensor can operate without interfering or being interfered with, even when in close proximity to other potentially highly integrated scientific instruments and satellite subsystems. As energetic particles are a byproduct of the reconnection energy conversion process, such measurements are of critical importance the MagCon mission.

Energetic charged particles are found throughout the Earth's magnetosphere. As noted in the science objectives, these particles often pose natural hazards to satellites: their extreme energies can lead to deep dielectric charging of sensitive electronic parts, or to intrinsic damage of microelectronics through ionizing radiation. Electrons, protons, as well heavy ions are found with energies well above the thermal plasma, extending to exceptionally high energies (many 10's to >1000's of keV), described typically by a power law velocity phase-space distribution. Many mechanisms have been proposed to explain how energetic charged particles are accelerated and transported to such high energies in response to substorms, including: diffusive, adiabatic transport; drift-resonant or betatron acceleration; wave-particle interactions; and through multiple encounters with inductive or strong parallel electric fields.

State-of-the-art energetic particle instrumentation on traditional spacecraft missions use multiple-element solid-state detectors, time-of-flight systems with large geometric factors, and/or high-voltage electrostatic or magnetostatic charged particle optics. These measurement implementations are mandated by the desire to measure with great accuracy and fidelity not only the full three-dimensional flux of the electrons and total ions, but also the mass, energy, and charge state of the individual ion species. These last requirements greatly drive the resources needed to measure the low fluxes of secondary ion components with complete pitch angle and high-energy-spectral resolution over a large dynamic range.

For any nano-satellite mission, such resources will simply not be available in the forseeable future using current technologies. Therefore the focus the Mag Con instrument shall be on electron and total ion measurements with modest, but adequate, pitch angle coverage and energy resolution and range. These should be feasible on a spinning, 10 kilogram spacecraft using advanced solid-state sensors and hybrid analog/digital electronics in a pin-hole type imaging system (e.g., such as that employed on the NASA Polar CEPPAD/IPS experiment).

While the technologies exist for constructing such a sensor (ion implanted solid-state detectors and hybrid, synchronous, multichannel analog to digital convertors), such extreme miniaturization has not been required and, therefore, has not been attempted. The Mag Con energetic particle instrument should be capable of measuring ~20 keV to >500 keV electrons and ions in at least eight logarithmically-spaced energy bins. This would probably be accomplished in two separate telescopes (one each for ions and electrons). In this type of particle sensor, clean ion measurements are usually obtained with the use of powerful permanent magnets that sweep electrons from the ion field-of-view owing to either the Lorentz force. Either sophisticated passive magnetic shielding, or differential correction of the electron and ion fluxes could be employed. Given these considerations, and based on previous similar designs, a mass resource of 0.5 kg and a power consumption of 0.5 Watts is envisioned. The telescopes should be mounted so as to provide as complete coverage in pitch angle as is possible during one spacecraft spin. For regions near the magnetic equator, this would mean having a spin axis pointed toward the sun.

Finally, to optimize systems resources, both the energetic particle sensor and the plasma instrument should be as fully-integrated with the magnetometer as is possible. On-board processors could be used to "despin" the magnetic field data on the fly, feed those data into the charged particle data streams, and extract relevant properties such as the fluxes parallel and perpendicular to the magnetic field and possibly Maxwellian or kappa-function fits to the particle velocity phase-space distribution function. Such a highly integrated package would allow for moments of the distribution to be stored and downlinked, rather than the much more voluminous individual data points. This represents a greatly reduced volume of telemetered data, while preserving critical scientific information. Such cross-instrument integration and synergy might be needed when considering resources for the communications component of the nanosatellite and could be a key technology element of the Mag Con stellation scenario.

SPACECRAFT

The Magnetospheric Constellation mission will be enabled by identical Nanosats which will each be released by a single deployership (or Dispenser Ship). The Deployership and it's cargo of Nanosats will be taken to the initial orbital position and attitude by a Delta II 7925-9.5 launch vehicle and then separated. It is envisioned that multiple nanosatellites will be released at perigee (3 Re.) Further multiple releases will be made at subsequent perigees (3 Re.)

DEPLOYERSHIP

The Deployer Ship preliminary requirements are given in the figure below.

Dispenser Ship
Preliminary Requirements

Revised 3-17-99

Size	88.6" diameter X 98.8 high
Shape	Cylindrical
Volume	9.66 E 4 cu in
Number of Probes	92 Strategy I
	Initial satellite spacing within each orbit TBD
Launch Vehicle	Delta II 7925A
Mission Orbits	All perigees = 3 Earth Radii (Re)
	Apogee 12 ER, 14ER, 16 ER etc.
Mission Lifetime	1 to 2 months
Radiation Environment	10 krads total dose over mission lifetime (scaled from Nano-sat mission Requirements) Latchup immune SEU = TBD LET = 90 (TBR)

Deployment	Delta places dispenser ship in a 1000 Km X 20 ER with an orbital inclination of 28° with a nodal crossing of 270° and a 0° argument of perigee. The Dispenser Ship's propulsion is used to change the orbit to 3ER X 20 ER with orbital inclination of 7.5°. The Nodal crossing and Argument of Perigee are not changed. With these parameters the line of apsides lies in the plane of the Earth's equator. The projection of the line of apsides on to the ecliptic points to the First of Aries. The angle the projection makes with the sun Earth line is dependent on date of launch. The angle is 0° when launch is in the autumn or spring and 90° in summer and winter.

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Inclination	Typically 7.5° from Earth equator initially
Orbit Periods	Dispenser ship orbit is 2.25 days (appx.)
Orbit Apogee Control	1 % of apogee radius
Orbit Position Knowledge	Science requirement: ±20 Km Communications requirement: Less stringent than the Science Requirements
Eclipse Duration	Less than 1 hour if launched between Oct. '08 and before Sept '09 with argument of perigee = 0° and nodal crossing of 270°.
Mass	Launch Mass 1476 Kg (Delta capability 1000Km X 20 ER X28° inclination) No margin (save those held by Delta)
Power	Less than 100 watts Batteries size for eclipse periods given above.

Power and Mass Allocation

	Watts	Kg
Power System	26	11.5
C&DH	27	5.6
ACS	10	6
RF Communications	11.2	12
Thermal	5	6
Propulsion wet	2	252
Structure	0	161
Payload (92 Nano-Sats)	0	920
Total	**81.2**	**1374.1**
Margin	18.8 (23%)	101.9 (22.4%){no margin for nano-sats}

Battery	40 to 70 cycle lifetime:	12 amp-hrs at 28 volts 30% max depth of discharge

Power Bus Voltage	28+/- 7 VDC
Thermal	heaters, insulation and coatings
Stabilization indicates that	Initial spin appx 70 RPM from launch vehicle (ACS analysis
	An initial spin rate of 40 rpm is preferable to hold thrust vector during orbit adjusts)
Attitude Control	Spin stabilized 20 rpm = 1/3 rev/sec (TBD see above) Spin rate knowledge: < $2x10^{-5}$ rad/sec Spin axis position knowledge: < 0.1° Spin axis drift rate: < 0.1° over 30 days Spin axis perpendicular to ecliptic within +/- 20 deg (non nanosat deployment mode)
Orbital Maneuvers	Flip 180° to align thrust vector Two burns to raise perigee and change orbital inclination
from 28° to 7.5°	
Sun Synchronization	Sun sensor Step rate = 2000 Hz (TBR) Resolution = 0.1°
Inertia	Izz/Ixx > 1.05
TM Data Rate	2 kbps
Data Storage	1 G bit on card in Processor Unit
Transmission Rate	Up to 100 kbps TBD
Command Rate	1 kbps
Communications:	
Transmit Power	1.5 watts RF output
Antenna	X Band patched. commutated, band antenna. 2 needed (1 top ; 1
bottom)	
Antenna Gain	> 0dB
EIRP	1.76 dBw
Tlm Frequency	8470 MHz requested (X-band) Lower preferred as long as antenna does not violate inertia
requirement	
Cmd Frequency	7209.125 Mhz (X-band)
Ground Terminal	11m antenna
G/T	35.4 dB/°K X band
Orbit determination	One-way doppler Stability = $5x10^{-8}$ per day

DEPLOYER SHIP SYSTEM DESCRIPTION

OVERVIEW

The Deployer Ship (DS) provides the initial orbit, attitude and spin rate for each of approximately 100 Nanosats. The orbit selected is 3Re perigee by 20Re apogee, in a 7.5 deg inclination (wrt the equator), and is achieved with a Delta Launch Vehicle and a DS-based propulsion system. Initial spin rate is 70 RPM at Delta LV deployment. The spin rate is 20 RPM at nanosat deployment. The Nanosats then use their own propulsion and guidance to reach their final orbits.

In order to fulfill these requirements, the DS uses a Payload Separation System provided by the launch vehicle, a Bi-propellant propulsion module, Power Module, Avionics/C&DH/Comm box, primary structure, Nanosat support-and-release structure, and requisite ancillary sensors and hardware for thermal and attitude control. The mission duration is 2 months. The Nanosats must each be released at perigee, this is driven in large part by the number of satellites to be released simultaneously and release timing.

The Structure is designed to a 1.25 Factor of safety on ultimate. Radiation shielding is used only on the 3 most sensitive components: C&DH, Comm and Power Supply Electronics.

SUBSYSTEM REQUIREMENTS

The Mechanical requirements are summarized in the following Table:

ITEM	REQUIREMENT
Overall S/C Structure (not including antenna, thrusters, Instrument protrusions)	Delta Launch loads Environment, CG and Frequency
Structure Weight	Approximately 200kg
Operating Temperature	-10 to +30 °C
Depolyership Spin Axis in Nanosat Deployment Mode	Spin Rate is 20 RPM; Spin Axis aligned to Perigee Velocity Vector to within +/- .5 deg

Deployership Spin Axis in Non Nanosat Deployment Mode	Spin Axis perpendicular to ecliptic within +/- 10 deg
Stabilization	Passively Spin-Stabilized to 0.1° cone-angle

SUBSYSTEM COMPONENT DESCRIPTION

The DS structure will be graphite composite. Components will be attached to the upper and lower decks through inserts bonded in the composite. Thermal conduction is assured through physical contact of the surfaces.

The DS Structure physically supports all components through launch and operational scenarios. It is a cylinder approximately 90" in diameter and 100" high. The 30cm diameter by 10cm high Nanosats are carried aloft in the support-and-release structure bays as shown in Figure 1. The structure is a combination of Aluminum and composite honeycomb.

The support-and-release Structure restrains the Nanosats during launch and deploys them at a precise time during perigee. The Release Mechanism rests almost entirely on the DS. It consists of a set of prongs protruding from the bays. The prongs fit into fittings embedded in the Nanosat vertices. Thus the Nanosat is clamped in place during launch. One set of prongs can pivot about an axis parallel to the DS spin axis, allowing simultaneous deployment and spin-up.

The Bi-prop system resides at the bottom of the DS, it provides delta-V through a 100lb-f nozzle and ACS through 1lb-f thrusters.

A Power System with a fixed Solar Array and 12 Ah Battery provides up to 100W.

The C&DH subsystem supplies the required attitude, thruster and release mechanism timing, and Communications operations.

Communications is achieved primarily at the 3 Re perigee through a Nanosatellite-heritage Omni antenna.

Function	Physical Description	Description
Structure	Cylinder with internal ribbing and composite Decks; External nanosat Bays	Honeycomb composite decks and ribbing w/graphite facesheet; Bays are aluminum strut
ACS	Passively Spin Stabilized; Initial Spin from Launch Vehicle	1 lb-f Nozzles Fed off Bi-Prop Delta-V system
Deploy Nanosat from DS	C&DH Command received by Release Mechanism	Prongs clamp S/C and provide simultaneous deployment and spin-up on command. Takes launch loads; interface is an embedded fitting. This component resides on and is commanded by the DS.
De-Orbit	Retrograde Burn	TBD (may not be necessary)

SUBSYSTEM PERFORMANCE

Launch loads are most severe, however margins will be adequate. Qualification loads will be applied during vibration testing. Substantial Analysis is needed to finalize design.

Pointing requirements achieved through Passive Spin control. Nanosats are deployed to preserve the inertia ratio spin stability of the Deployership. Active control may be necessary due to Nanosat non-deployment.

Communications is easily achieved through omni system.

Spin balance and Mass Properties are within requirements.

SUBSYSTEM TRADE STUDY CONFIGURATIONS EVALUATED

Material: An all-Aluminum structure would be easy to manufacture, and provide radiation shielding for internal components, but would be heaviest. A combination structure is currently baselined

Use Nanosat Solar Arrays for Power: Eliminates need for separate Solar Array. Available power decreases as nanosats are deployed. Need robust electrical umbilical connector. Fixed DS S/A currently baselined (low mass).

X vs S-band: S-band availability is tight. X-band would simplify system. Current Baseline is X-band only.

Release Mechanism (3, 4 or 5 contact points; linear vs. torsion release spring): Detailed analysis is required to determine number and locations of attach points. Linear spring might imply cup-cone attachment.

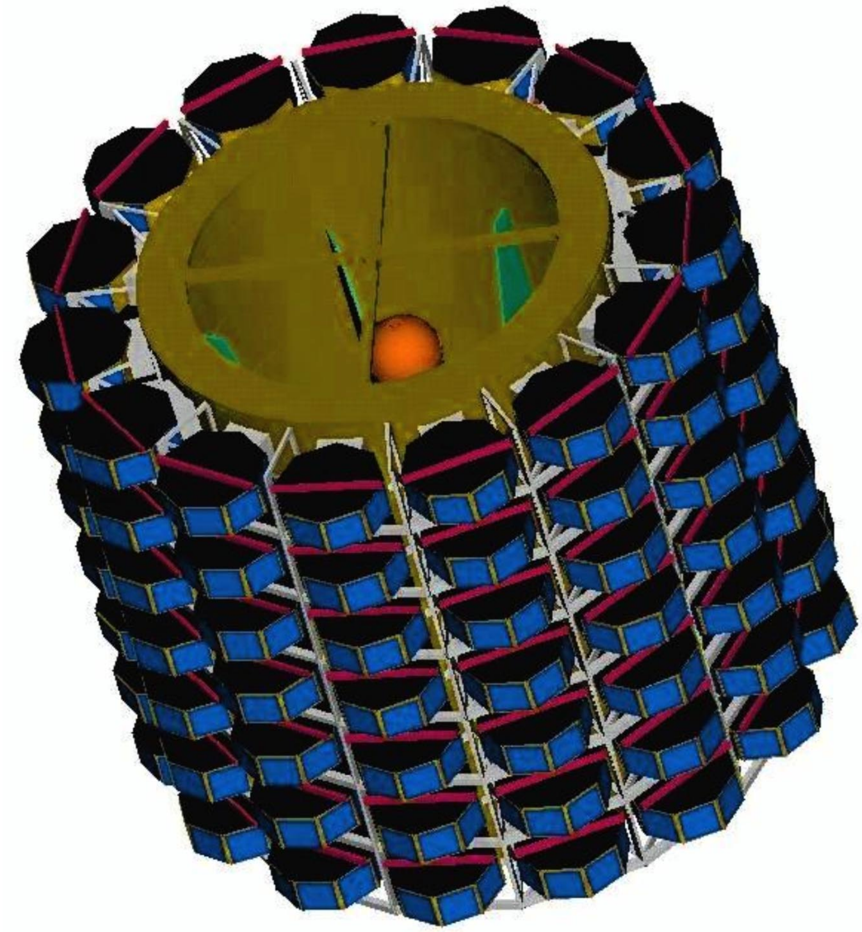


Figure 1. Deployer Ship

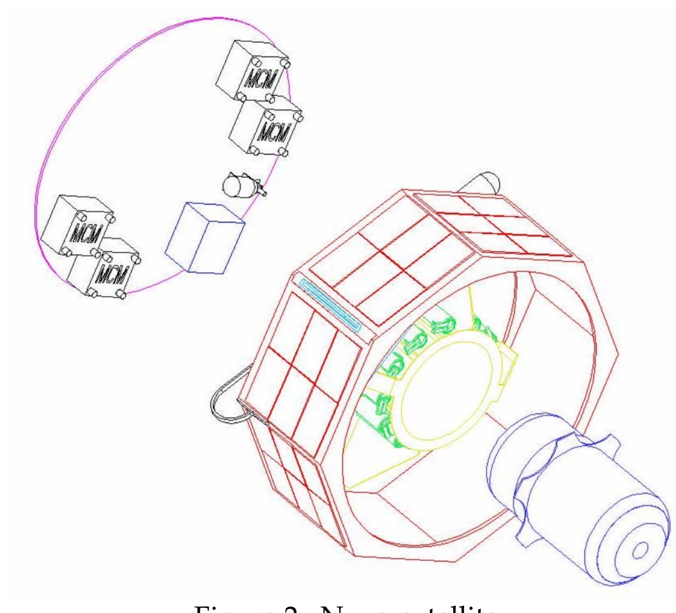


Figure 2. Nano-satellite

NANOSATELLITE

As was stated earlier the Magnetospheric Constellation mission is enabled in terms of constellations of spacecraft by advanced technology. it is constrained in terms of the number of Spacecraft within the constellation by the size & mass of each Nanosat & the Deployer Ship. The capability of the Delta II Launch Vehicle to Place the Deployer Ship & it's cargo of Nanosats into orbit is a key factor. In order to populate the Magnetospheric Constellation with the greatest number of Nanosat and still fit within the fairing of the 7925-9.5 Delta Launch vehicle, a total of approximately 100 Nanosat was sized for the mission. Each Nanosat is defined as 10 kg as shown in Figure 2. In order to meet the size and weight constraints of the Nanosats advanced technology has to be employed. This section will describe the Nanosat and it's subsystems.

MECHANICAL SUBSYSTEM

SUBSYSTEM OVERVIEW

The Mechanical subsystem is comprised of a Structural Bus, Release Mechanism for deployment from the Deployership, and the deployable Instrument Boom.

The Structure physically supports all components through launch and operational scenarios. It is an octagonal prism 30cm across the flats and 10cm high. The Bottom Deck is composite honeycomb; the side-walls which support body-mounted solar arrays are made of a thermally isolated, filament-wound composite shell. The Top Deck has high thermal conductance, and is of honeycomb or ribbed facesheet. The precession thruster, Ultra-Stable Oscillator and one Instrument Detector are mounted on the Bottom Deck. The remaining boxes (including C&DH, Instrument, Battery) are mounted on the Top Deck for improved heat dissipation. The Solid Rocket Motor and Comm Antenna are mounted to bottom and top decks respectively and are aligned to the spin axis.

The Release Mechanism resides almost entirely on the Deployership. It consists of a set of prongs protruding from arms sticking out from the Deployership. The prongs fit into fittings embedded in the S/C vertices. Thus the S/C is clamped in place during launch. One set of prongs can pivot about an axis parallel to the S/C spin axis, allowing a simultaneous deployment and spin-up of the S/C.

The Magnetometer is deployed on a stacer-type Boom in the radial plane of the S/C CG.

The Structure is designed to a 1.5 Factor of safety on ultimate. Radiation shielding is used only on the 3 most sensitive components: C&DH, Comm and Power Supply Electronics.

SUBSYSTEM REQUIREMENTS

The Mechanical requirements are summarized in the following Table:

ITEM	REQUIREMENT
Overall S/C Structure (not including antenna, thrusters, Instrument protrusions)	10cm high by 30cm diameter; Magnetically "clean;" minimize sharp corners; passively spin-stabilized
Weight	2.5 kg
Operating Temperature	-80 to +50 °C
Nanosat Spin Axis Attitude During Deployment from Deployership	Spin Axis Aligned to Perigee Velocity Vector to 1 deg (0.5 deg from DS + 0.5 deg from Nanosat)
Nanosat Spin Axis Attitude During Free Flight	Spin Axis Perpendicular to Ecliptic to within +/-.5 deg; 0.1 deg Cone Angle
Science Mode Spin Rate	20 RPM
Launch Loads	10G in X, Y, Z; GEVS Qual-level Random
Frequency	60 Hz axial; 30 Hz lateral
Acoustic	Delta II Levels

SUBSYSTEM COMPONENT DESCRIPTION

The structure for the Mag Con Nanosatellite will be graphite composite, cast aluminum, injection-molded composite or a combination of these. Components will be attached to the upper and lower decks through inserts bonded in the material. Thermal conduction is assured through physical contact with the surfaces.

Function	Physical Description	Description
Structure	Top & Bottom Decks w/Octagonal Shell	Monocoque; Filament-wound shell for sidewalls provides high stiffness, large thermal variation and low weight; shell could incorporate structural battery. Top deck removable for I&T.
Deploy S/C from DS	Opposing rotatable "Prongs"	Prongs clamp S/C and provide simultaneous deployment and spin-up on command. Takes launch loads; interface is an embedded fitting. This component resides on and is commanded by the DS. Spin attitude maintained passively by balanced component placement.
Thermal Control	High Conductance Panel	Chemical-Vapor-Deposited Diamond facesheet on honeycomb core provides superior heat dissipation; Moderated by blankets for eclipse period.

SUBSYSTEM RESOURCES REQUIRED

Subsystem	Interface	Purpose/Amount
Power	Power Harness	Mag Boom Deployment/5W for 2

		min
Deployership	Release Mechanism	S/C Release/Timing/20W for 0.1sec
C&DH	Power Harness	Boom Deployment Command/Timing

SUBSYSTEM PERFORMANCE

Launch loads are most severe, however margins will be adequate. Qualification loads will be applied during vibration testing.

Spin balance and Mass Properties are within requirements.

1. SUBSYSTEM TRADE STUDY CONFIGURATIONS EVALUATED

Structure:

A. Material: An all-Aluminum structure would be easy to manufacture, and provide radiation shielding for internal components, but would be heaviest. A Graphite-composite structure is currently baselined. A thermo-plastic structure would be lighter, and cheaper to make in the long run especially as a path-finder for the 100-plus S/C Mag-Con Mission. Cast Aluminum or alternate composite may save weight and fab cost in the long run, once non-recurring costs have been incurred.

B. Central Spine: Graphite Composite baselined due to good heritage. Spine may be needed if Top Deck cannot support loads.

Release Mechanism:

A. Clamp loads are carried either through a central spine or through top and bottom decks alone. The latter is more space-efficient, however analysis must be performed to verify top deck can take those loads. It is advantageous to keep the Top Deck as a highconductive material, which is not as strong, necessitating some load-sharing, such as through a central spine.

B. (3, 4 or 5 contact points; linear vs. torsion release spring): Detailed analysis is required to determine number and locations of attach points. Linear spring might imply cup-cone attachment.

Mag Boom:

(Telescoping vs. Folded Boom): Telescoping allows rigidity and ease of placement on CG plane, but may not be available in non-magnetic material.

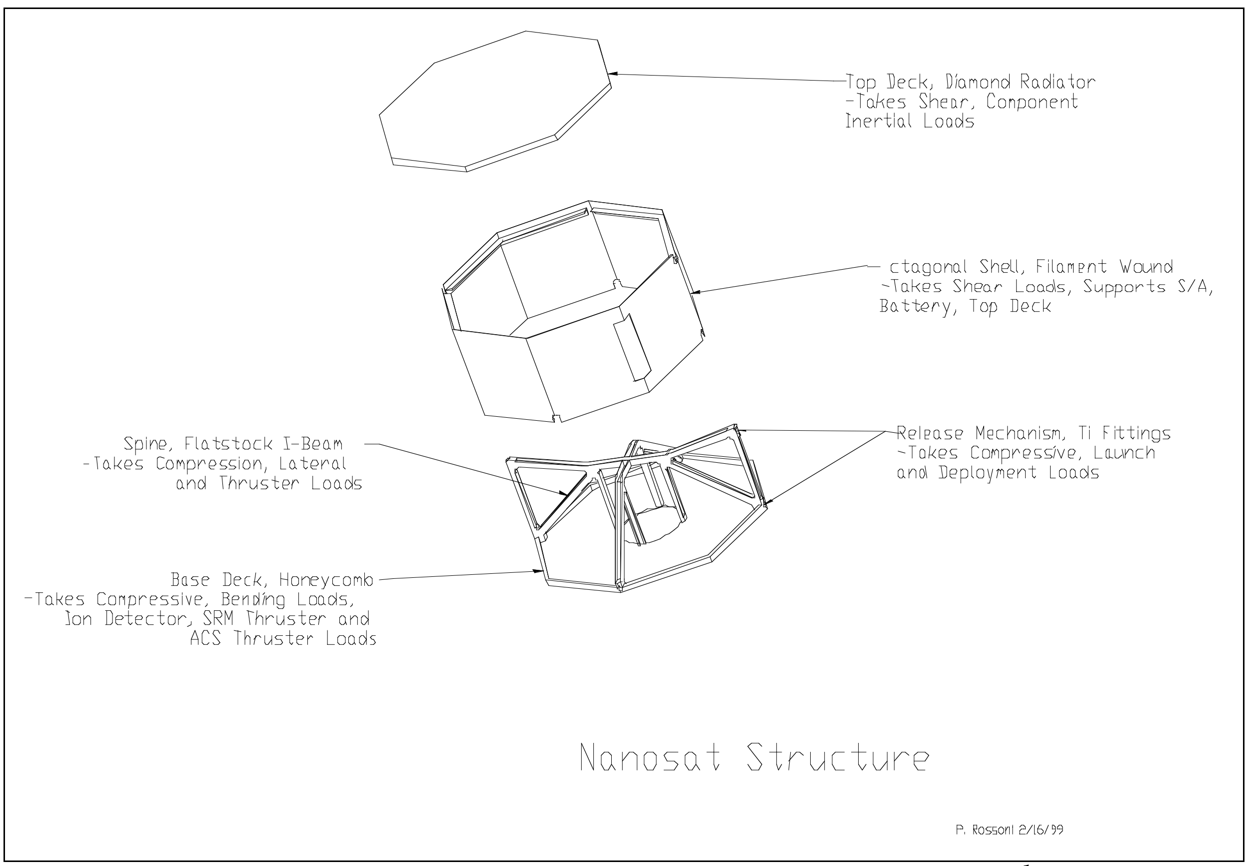


Figure 1. Nanosatellite Structural Subsystem. Deployable Instrument not Shown

COMMAND & DATA HANDLING

Nanosat C&DH Description

REQUIREMENTS

The requirements for the Command and Data Handling (C&DH) subsystem for Nanosat are as follows:

ITEM	REQUIREMENT
Power	0.8 Watts
Weight	0.25 kgrams
Radiation	100k Rad Si (Total Dose)
Data Rate From Instrument	2 kbit/sec
Data Storage	2 Gbits
Uplink/Downlink Protocol	CCSDS
Uplink Rate	1kbits/s
Downlink Rate	256kbits/s

TECHNOLOGY

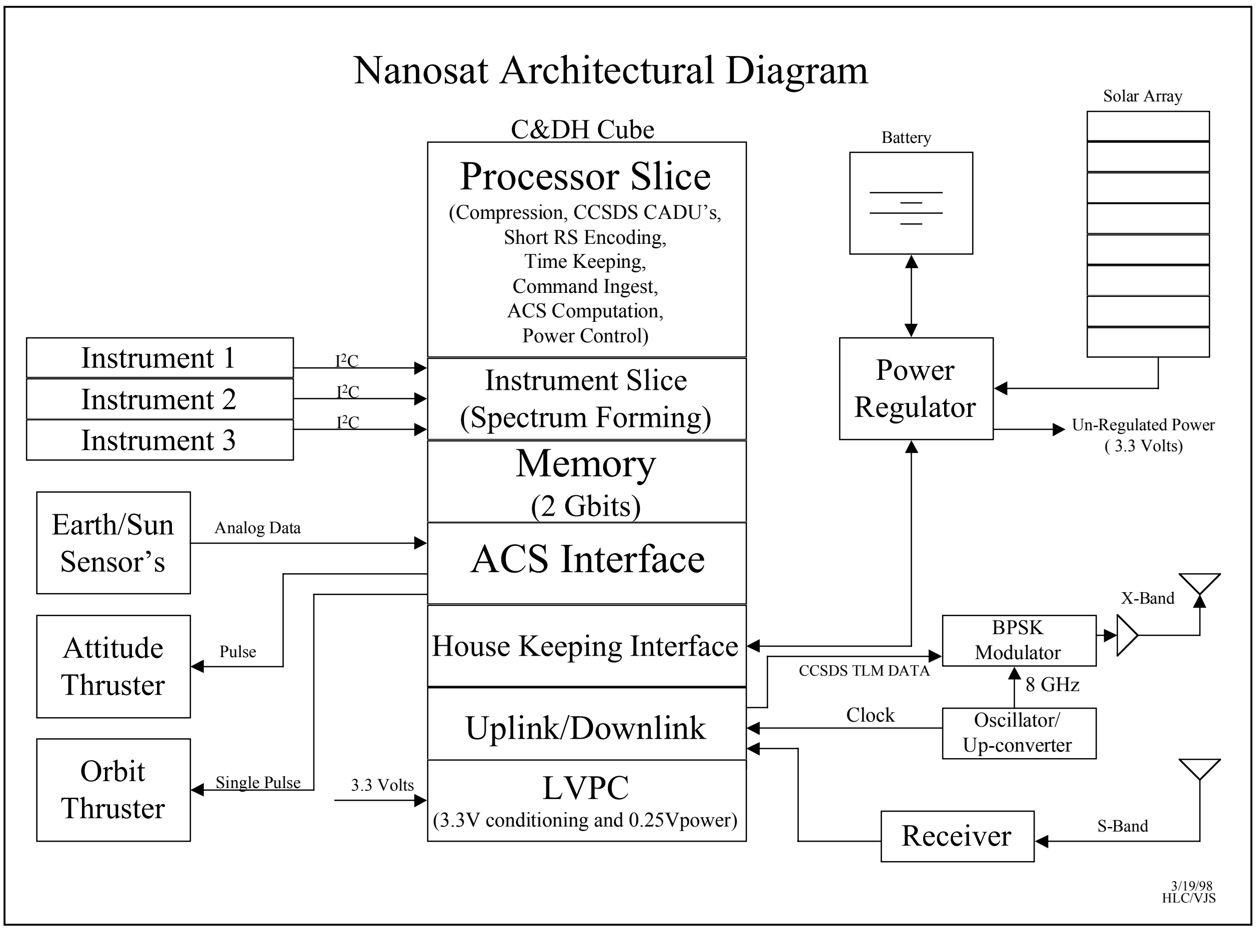
In order to meet the challenges imposed by the requirements of Nanosat, technology that is not currently available must be developed. The first technology that needs to be developed, to meet the weight requirement of 0.25 kgrams is an inexpensive lightweight packaging technique. This will be accomplished using a multi-chip module (MCM)technique described in the packaging section below.

To meet the power requirements a combined effort in the reduction of mass, power, size, and cost is underway to produce optimal electronics. The CMOS Ultra Low Power Radiation Tolerant (CULPRiT) system on a chip, and "C&DH in your Palm" are technologies that will enable the power reduction required for nano-satellites. The goals of these technologies are a 20:1 power reduction over current 5 volt technology, foundry independence of die production, and radiation tolerance.

Every three years memory technology enables a doubling of memory capacity and a halving of silicon area. Memory trends starting in 1996 are toward a 3.3 V core and a 3.3 V I/O, reducing by 1/3 the power for Gbit size solid state recorders. Trends in packaging technology are enabling denser 3-D stacking in smaller volume packages for multi-bit stacks in the next three to five years. This will be accomplished by incorporating Chip Scale Packaging technology where the package is less than 1.2 times the area of the silicon. DRAM memory will be at the 128 Mbit per die level within the next three years. With these current trends, it appears promising that an off-the-shelf solution is viable for the C&DH subsystem of a nano-satellite.

ARCHITECTURE

The Nanosat C&DH Architecture will incorporate CULPRiT devices in a multi chip module (MCM) Cube. The stacked architecture is shown in the attached diagrams. The C&DH subsystem is made up of 6 functional slices of the cubic stack. The first slice is the processor slice which will process commands, make data packets, perform the timekeeping functions, perform ACS computations and control power. The second slice is the instrument slice. This slice will interface to the instruments via an I^2C bus. This slice will also manipulate the data into the required format. The Third slice is a memory slice (or slices) which will contain 2 Gbits of SRAM or DRAM. The Next slice is the ACS Interface slice which will interface to the Earth/Sun Sensor (Analog data), Attitude Thruster (Pulse command) and Orbit Thruster (Single pulse). The fifth slice is the House Keeping Interface, which interfaces to the Power Subsystem electronics and reads other analog telemetry such as thermistors, voltage and current monitors. The last card is the Uplink/Downlink card. This card interfaces to the transmitter and receiver. This card sends CCSDS packet telemetry to the ground and receives CCSDS packet commands and processes them.



FLIGHT SOFTWARE

GENERAL

Software embedded in the flight computer/s will perform control functions for the spacecraft. The software function can be resident in a centralized computer or be highly distributed. The software will enable as much onboard nanosat autonomy as is feasible. With a constellation of as many nanosats as is needed for the Magnetospheric Constellation mission and the limited ground station coverage/capability (for technical & cost reasons) an autonomy function is desirable. The flight software will have to be very capable to handle routine spacecraft bus and instrument operations. It will also have to be sophisticated in handling contingency mode operations upon detection of anomalous operation in any of the spacecraft subsystems or instruments. As a goal, it is envisioned that many , if not most, of the software functions previously performed for spacecraft, on the ground, will be performed on-board the spacecraft. The table below provides some examples of the Nanosat flight software functional capability.

SUBSYSTEM/FUNCTION	EXAMPLES
OperationalMode	Operate S/C; Safe S/C
Mechanical	Deployment
C & DH	Telemetry, Commanding
Instruments	Operations & Data Processing
Thermal	Louver/Heater Control
Power	Regulation; Charge /Discharge
RF Communications	Gnd Station Data Dumps
Propulsion	Time of Firing
Attitude Determination & Control	Sensor processing; Control Software; Reorientation

	maneuver calculation

AUTONOMY

The objective of the nanosat autonomy effort is two-fold:
1. Maximize the scientific return given the limited TM resources.
2. Minimize the need for ground-based control of the constellation.

The nanosat baseline autonomy concept is summarized below.

Intelligent Science Return
Data acquisition rates for earth-orbiting missions are projected to exceed terabits/sec. The resources required to store and transmit the resulting data sets would increase the size and mass of onboard systems. Furthermore, scientific return could be compromised as only a fraction of the massive volumes of data telemetered by instruments may be analyzed because of limited data analysis budgets. To address this issue, the onboard remote agent will use heuristic techniques to classify events. Instead of viewing data from the spacecraft sensors and instruments as independent streams, the heuristic system will analyze the full complement of inputs to assign degrees of interest to the science data. High interest data will be stored and low interest data rejected. This intelligent filtering will thus reduce storage, RF and ground analysis requirements.

Instrument Control
The overarching objective is to maximize the science content of the data stream. Classical compression and event identification algorithms and yet to be developed pattern recognition methods are among the implementation methods under consideration. This area will be addressed in more depth with the science working group. Instrument operation will also be monitored to detect and correct scenarios which are potentially damaging to the instrument , e.g. excessive particle count rates or corona from the high voltage system.

Control of Spacecraft Subsystems
Nanosatellites in distant orbits are out of communications range of a ground station for nearly a week. Spacecraft subsystems could be compromised if faults occurring during this blackout period were not readily addressed. An unacceptable loss of scientific data could also occur. Therefore, the onboard agent will incorporate the capability to

detect, diagnose and recover from faults. Corrective actions will be consistent with a maximum-science goal: the actions taken will attempt to maximize scientific return, not necessarily maximally protect the spacecraft.

Ground Function

Certain failure scenarios may not be correctable by the onboard agent. These faults will be deferred to the ground station for handling. Each spacecraft will include in its telemetry stream data on the health and status of each subsystem and a history of commands autonomously issued since the last ground contact. The ground system will then attempt to diagnose problems based on this data. Since the ground agent will have greater processing power than the onboard agent, the probability of resolving faults is higher. Additionally, within the ground system will reside collective knowledge of actions taken by all satellites in the constellation by virtue of the dumps made during each contact. From this data the agent can detect trends and systemic conditions not otherwise observable onboard the spacecraft. Once corrective actions are determined, they can be implemented by uplinking commands or by modifying the onboard agent. Scenarios that cannot be addressed by the autonomous systems will be deferred to a human operator.

Power

The major objectives are deriving and optimizing the state of the battery, detecting faults and attempting to recover from them. Although the power system is single-string, options which require minimal resources are available to correct or isolate certain types of faults. We plan to develop a power system testbed in FY99 to investigate various methods of controlling the power system. Among the targeted methods are “classical” control as well as model-based reasoning using agents, fuzzy-logic, and heuristics. Since the power-system modes controlled by S/W are low-bandwidth in nature, the average CPU cycles required will not be significant.

GN&C

Proper orbit insertion is dependent on the deployer-ship and the motor burn. If a solid motor is used, only the ignition is controllable. Attitude control will be derived from the sun sensor, earth sensor, the gas thrusters and heritage CPU-resident algorithms. An effort is in progress to determine if orbit control can also be computed using these sensors.

RF Comm

If the orbit is not determined onboard the nanosatellite, the transmitter must be commanded, in some manner, to turn on near perigee. If the orbit is determined onboard, autonomous operation of the s/c transmitter would be possible.

Thermal

No ground or onboard resources are required if a passive system is used. However, if the Capillary Pump Loop (CPL) option is onboard resources will be required to start the system, and to react to CPL deprime.

Challenges

These highly autonomous systems will present a unique set of challenges not only to the system designers, but also to those involved in the test of the spacecraft. Careful consideration must be given to the design of the test program to ensure that the entire state-space of the remote agents is validated and verified. It is equally important to implement this program in a cost-effective manner. However, we could likely justify exerting considerable resources to address this issue since the methods developed to solve these challenges can be applied to numerous missions.

ATTITUDE AND ORBIT DETERMINATION & CONTROL

The Magnetospnheric Constellation mission has the capability to perform Attitude Determination and Control for the individual nanosat in the Constellation throughout the mission. The mission will also perform Orbit Determination but aside from placing each nanosat in it's initial elliptical orbit there will be no further orbital control. This is necessary because of the limited propulsion system onboard each spacecraft due to the total mass constraint for enabling the mission.

Attitude Determination & Control

The requirements for each nanosatellite are based on the simple spin stabilization chosen as baseline. Miniature sun and earth sensors will be employed on each spacecraft. The miniature precision "fan" type sun sensor will locate the sun on the celestial sphere during every spacecraft rotation. The sun sensor will weigh approximately 0.25 kg and operate at 3.3 Volts @ less than 0.1 watt. The required resolution which will be met is 0.1 deg. The miniature earth sensor also will have a "fan" type field of view. The pointing accuracy is 0.05 deg from 3 Earth Radii (ER). The power and weight requirements of this sensor are 0.1 Watt and 0.2 kg.

Attitude Control will be enabled by a determination of the attitude via the sensors mentioned above along with the desired a priori attitude. The difference of the two is the required reorientation maneuver. A reorientation maneuver will subsequently be undertaken as required by either a low power hydrazine thruster or a low power miniature cold gas thruster. Both of these candidate propulsion systems technologies are presently being considered for the mission. If the cold gas thruster hardware is chosen for the mission then a small nutation damper will be employed to damp out residual nutation following the attitude maneuver. This will not be necessary for the hydrazine system since the hydrazine in it's tank would naturally damp out the residual spacecraft nutation. □□□□□□□□□□□□

Orbit Determination

The Nanosat Program has identified a number of orbit determination concepts which are considered appropriate for the mission. The following table denotes the mission orbit requirements.

ORBIT	REQUIREMENT
Control	+/- 0.5 Re @ each apogee
Knowledge	
Science	+/- 20 km
Communications	+/- 60 km

The orbital control requirements in the above table are placed on the Nanosat for initial orbital positioning upon separation of the spacecraft from the deployer ship and are not used for stationkeeping of the spacecraft at subsequent apogees.

Basically, there are four methods under consideration to perform orbit determination. The ultimate determinant as to which method gets selected depends on the satisfaction of the above requirements and also meeting the small physical constraints posed by the nanosat (i.e. weight, volume, power) The four concepts currently under investigation to perform this function are:

<u>Magnetometer methodology</u>
The onboard magnetometer is used to measure the earth's magnetic field in a low altitude region (around perigee.) This data is then compared to a map of the field which is stored in memory. A Kalman filter propagates the ephemeris for a complete orbital solution.

TDRSS Onboard Navigation System (TONS)
The doppler shift derived from the TDRSS communication signal is used to generate nanosat orbital data in this method. The concept as applied to the highly elliptical orbit of the Nanosat and also it's impact on the primary physical characteristics of the spacecraft (weight, power, volume) will have to be analyzed during the following phases of the Mag Constellation mission so as to insure that they do not become adverse mission drivers.

Global Positioning System
This System will also be investigated for orbit determination. "GPS on a chip" technology will be studied closely for employment on this mission.

Ground Based Orbit Determination System
A ground beacon at the operational control station can be used to transmit a signal to each nanosat for onboard orbit determination. The impacts to the operational control station and to the nanosat onboard C & DH subsytem and it's packaging will be studied in future phases of the mission prior to design

PROPULSION SUBSYSTEM

Each nanosat will require a propulsion system. The spacecraft have Delta-V (velocity) and attitude control requirements which will have to be satisfied by the propulsion subsystem. The Delta-V requirements are necessitated by the need to raise and/or lower the nanosat apogee from the deployer ship orbit. Attitude control requirements are necessitated by the need to reorient the nanosat spin axis from the velocity direction to the science direction. Existing propulsion systems will not satisfy the requirements for the nanosat spacecraft. Specific technology developments in the propulsion area are being undertaken to satisfy these requirements. The propulsion system requirements are found in the table below.

Delta-V Requirements

ITEM	REQUIREMENT
Total Impulse	3000 – 7000 N - sec
Thrust	445 N (max)
Input Power	<1W
Isp	280 sec

Attitude Control Requirements

ITEM	REQUIREMENT
Total Impulse	4 N-sec
Minimum Ibit	0.044 N-sec
Response Time	< 0.005 sec
Pulse Rate	1 Hz
Isp	60 sec

Figure- Propulsion System Requirements

Derivation of nanosatellite propulsion requirements

The current propulsion subsystem configuration for the nanosats consists of the following:

Orbit raising – Nanosat Solid propellant Motor (NSM); for primary delta-V propulsion needs, projected Isp = 280 sec., average thrust = 267 N

Attitude control – Miniature Cold Gas Thruster (MCGT); for re-orientating the spin vector, projected Isp = 60 sec., nominal thrust = 0.45 N

The following explains the derivation of nanosatellite propulsion parameters.

<u>I. Reorientation Thruster Requirements.</u>

It is assumed that the reorientation thruster is responsible for reorienting a 10-kg nanosatellite by 90 degrees. In order to ensure adequate attitude accuracy, the thruster should be sized such that approximately 100 pulses (one pulse per revolution) are required to perform the reorientation maneuver.

If the 10 kg mass of the nanosat is evenly distributed within a volume 30 cm in diameter and 10 cm in height, the moment of inertia about the spin axis is

$$Izz = 1/2 \times (10 \text{ kg}) \times (0.15 \text{ m})^2 = 0.1125 \text{ kg*m}^2$$

The spin rate of the nanosat was assumed to be 30 rpm for the purposes of this analysis (the actual spin rate had not yet been determined).

Therefore the magnitude of the angular momentum of the nanosat is

$$H = (0.1125 \text{ kg*m}^2) \times (3.14159 \text{ rad/s}) = 0.35 \text{ N*m*s}$$

and the magnitude of the momentum change is

$$(\text{delta } H) = H \times \alpha = 0.55 \text{ N*m*s},$$

where α is angle of rotation of spin vector in radians.

The moment arm of the thruster is assumed to be equal to the radius of the nanosat, or 0.15 m. Therefore the total required impulse from the reorientation thruster is

$$Ft = (0.55 \text{ N*m*s}) / (0.15 \text{ m}) = 3.67 \text{ N*s}$$

Each thruster pulse is assumed to be 0.100 seconds long. Therefore, for 100 pulses, the total thruster on-time is 10 seconds. Consequently, the required thrust of the reorientation thruster is the following:

$$F = (3.67 \text{ N*s}) / (10 \text{ s}) = 0.367 \text{ N} = 0.083 \text{ lbf}.$$

This number was rounded up to 0.1 lbf, or 0.445 N. Doing so allows ready comparison to state-of-the-art 0.1-lbf thrusters.

<u>II. Delta-V Thruster Requirements.</u>

The delta-v thruster was sized according to the following mission scenario, which does not necessarily reflect the current baseline mission. The selection of parameters is somewhat arbitrary.

1. Launch vehicle provides deployer ship with an orbit of 185 km perigee altitude, 7.62 Re apogee radius (one Re above GTO), and 28.7 degrees inclination.
2. Deployer ship raises perigee from 185 km altitude to 4.0 Re radius while reducing inclination to 7.5 degrees.
3. Nanosat Delta-V Thruster raises nanosat apogee to final mission radius (between 12 and 42 Re).

In reality, the initial separation apogee should have been only 6.62 Re, and the final perigee should have been only 3 Re. However there was some confusion about whether the specified orbital parameters had been expressed in terms of radius or altitude.

At any rate, the minimum-energy and maximum-energy nanosat burns require the following delta velocities:

minimum-energy, 7.62 Re to 12 Re: 314 m/s
maximum-energy, 7.62 Re to 42 Re: 814 m/s

Assuming that the initial nanosatellite mass including all propellant is 10 kg, and the specific impulse of the thruster is 280 seconds, the following propellant quantities and total impulses are required:

minimum-energy: 1.08 kg propellant; 2970 N*s total impulse
maximum-energy: 2.57 kg propellant; 7049 N*s total impulse.

Therefore the total impulse requirement for the delta-v thruster was set at 3000 to 7000 N*s.

Note that, if the nanosat perigee is set to 3 Re instead of 4 Re, the delta-v required for a given apogee change actually decreases slightly. Further calculations show that a propulsion subsystem delivering 7000 N*s is capable of raising a 3 x 7.25 Re orbit to 3 x 42 Re.

High Performance Technology Development

Solid Rocket Motor (Delta-V)

A solid motor is not a novel idea but it does not exist in the weight, size and power requirements that are necessary for the nanosat in order to implement the Magnetospheric Constellation mission. An initial study has identified technology drivers as the highly integrated spacecraft bus design; an acceptable Safe/Arm/Ignition System; and the cost, propellant mass fraction, and impulse accuracy balance. A systems approach for a low power and low mass Safe/Arm/Ignition system has been identified and the development is underway.

The technology goals of the solid rocket motor development are found in the following table.

ITEM	GOAL
Accommodate Any Impulse	3000 – 7000 N - sec
Max. Thrust	445 N
Isp	>/= 280 sec
Propellant Mass	80% of total motor inert wgt
Impulse Error	<0.5% max
Ignition Power	< 1W
Operating Temperature	-10 C to + 40 C
On – Orbit Storage	2 years (pre- firing)

FIGURE – Solid Rocket Motor Goals

Attitude Control Propulsion System

There are two technology development approaches being considered for the satisfaction of this requirement. One is a low power miniature cold gas thruster approach and the other is a low power monopropellant hydrazine thruster approach. This latter approach has the potential to satisfy both delta-V and attitude control system requirements. These two approaches are discussed below.

Low Power Miniature Cold Gas Thruster

This approach takes the form of a simple blowdown cold gas system for re – orientation maneuvers. Existing valve technology will not meet requirements. Technology drivers associated with this approach are low power/voltage coupled with high inlet pressures and response times. In addition there are leakage and cost issues which will be addressed in future development/studies.

The following table provides the technology goals associated with this concept.

ITEM	GOAL
Power	< 1 W
Voltage	3.3 +/- 0.4 V
Minimum I - bit	44 mN-sec
Pressure Range	100 - 1000 psi
Response Time	5 msec
Leakage Internal External	 < 1x10–4 sccs He (1x10-5 goal) <1x10-6 sccs He
Pulse Frequency	>/= 1 Hz
Cycle Life	1000
Flight Mass	50 g
Isp	60 sec

FIGURE – Cold Gas Thruster Approach

Low Power Monopropellant Hydrazine Thruster

This alternative to the cold gas thruster features a unique valve approach. The technology driver is the low power approach coupled with the required response time. The table below provides the goals associated with this alternative approach.

ITEM	GOAL
Average Vacuum Thrust	0.445 N at 250 psi
Minimum I - bit	44 mN - sec
Inlet pressure	100 - 400 psi
Power	1 W max

Voltage	3.3 +/- 0.4 V
Duty Cycles	0.100 sec on/30 sec off (0.3%) 0.100 sec on/3 sec off (3%) 60 sec on (steady state)
Response Time	30 msec to 90% thrust
Isp	220 sec

FIGURE – Hydrazine Thruster Approach

Other Possible Nanosat Technologies

Aside from the technologies baselined above the project is actively investigating the following encouraging developments.

A. Solid propellant Micro Electro Mechanical System (MEMS) based thrusters. This work is being undertaken at the NASA/Glenn Research Center. It is applicable to Attitude Control type maneuvers.

B. Advanced Monopropellants. This would entail a potential partnership with the NASA/ Glenn Research Center and a commercial vender. It is applicable to Delta-V and ACS type maneuvers.

C. Solid Gas Generator for cold gas thrusters. This has applicability to ACS applications

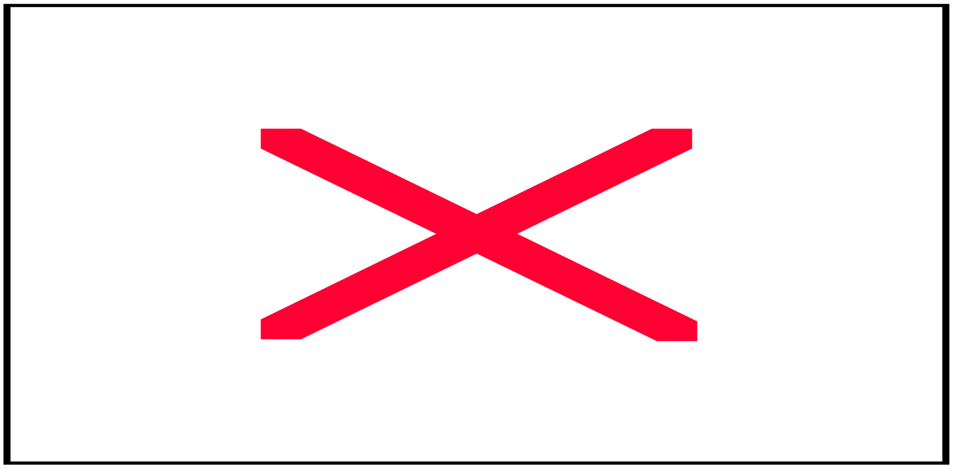
It should also be noted that other NASA/GSFC missions will require micro – impulse capabilities other than the Magnetospheric Constellation mission so that the propulsion capabilities described above and other possibilities are under intense scrutiny by the propulsion community.

POWER SUBSYSTEM

NANOSAT EPS SUBSYSTEM

SUBSYSTEM OVERVIEW

The electrical power system will be comprised of triple junction (TJ) GaAs solar cell on eight (8) flat panels of dimensions 11.5 cm x 10 cm each for converting solar power to electrical power and Lithium Ion (LiIon) batteries for energy storage. The Electrical Power System (EPS) delivers an average power of 4.0 watts to the spacecraft and instrument loads. Beginning of life (BOL) Max power available from the solar array is 7.0 watts with an end of life (EOL) at 5.71 watts at a 3 year end of life. The energy storage is two 1.2 ah LiIon batteries sized to handle the peak power of 4.39 watts or a option of handling a survival power of .711 watts for a 1.17 hour eclipse at a 60% depth of discharge (DOD). The driver for the solar array size is the condition where average power of 4.0 watts is supplied to the bus while the batteries are being charged from a 1.17 hour eclipse. Batteries can be used to augment the solar array during non-eclipse periods when the transmitter is operating. A margin of 25% is included in the load analysis.



NANOSAT LOAD ANALYSIS

This load analysis was done keeping the height of the spacecraft to 10 cm. The reserve row shows the excess power, which is extra contingency. Solar cell area can be reduced, or the use of lower cost Dual Junction cells can also be explored.

Survival	Average			Average		
	Day, Watts		Ni, Watts		Pwr	
Magnetometer	0.200		0.200		0.060	
Ion/Electron Analyzer		1.000		1.000		0.300
Instrument C						
Instrument D						
Total Science Loads	1.200		1.200		0.360	
ACS	0.300		0.300		0.075	
S/C CPU	0.800		0.800		0.200	
Propulsion	0.001		0.001			
Communications		0.063		0.500		0.070
Power	0.020					
Structural						
Thermal						
Reserve		0.650		0.650		
Total Spacecraft Loads		1.834		2.271		0.351
Total Power	3.034		3.471		0.711	

SOLAR ARRAY SIZING

Worst case eclipse:	70 min for every 70 days.
Array temperature:	30 deg C

Solar Array size @ 1350w/m**2: 0.0315 m**2 0.34 ft**2

Efficiency:	MJ GaAs BOL 0.22
	MJ GaAs EOL 0.18
Mission Life:	3 yrs
Power Output:	BOL 7.0 watts
	EOL 5.7 watts

BATTERY SIZING

At this preliminary stage, battery sizing is as follows for two different options:

Option #1: Using Survival Power to size the batteries.

Storage Requirement:	1.1 whr
Ampere requirement:	0.3 ahr
Depth of discharge limit:	60%
Number of Batteries:	1
Battery Ah rating:	1.2 ahr
Eclipse power:	0.711 w

Option #2, Using Night Power to size the batteries.

Storage Requirement:	6.7 whr
Ampere requirement:	2.0 ahr
Depth of discharge limit:	60%
Number of Batteries	2
Battery Ah rating:	1.2 ahr
Eclipse power:	4.338 w

NANOSAT EPS TECHNOLOGY DEVELOPMENT

A wide set of technology was reviewed for the Nanosat project, which included, flywheels, ultra capacitors, LiIon cells, single junction and dual junction GaAs solar cells. Because of the spacecraft size and mission requirements, the best technology will be dual junction GaAs and LiIon cells with a power bus below 5 volts.

1. Technology for the GaAs solar cells and LiIon cells already exist, however a 3.3 volt bus development will be necessary.

GaAs solar cell development. PanAmSat Corp, PAS-5 spacecraft was launched 27 August 1998 is flying dual junction GaAs cells. Multi junction cells are the next step and several LeRC and commercial technology efforts are under way. These cells are available today through Spectrolab and perhaps other vendors.

LiIon battery cell development. Several development efforts are under way including a GSFC effort by code 563. This effort procures domestic Lithium cells for space qualification testing for MAP and similar missions. Three vendors are involved in this effort: Mine Safety Applications Company, SAFT, and Yardney Technical Products, Inc. A plastic “flexible” LiIon cell is presently being developed by APL. Several other development efforts are underway through LeRC. This type of battery is available in the commercial markets, however space qualification is needed.

Low voltage power system development is a technology area that needs further study. There is no known development in this area and the Nanosat requirements suggest a 3.3 volt regulated bus. This bus technology effort would derive the best architecture of solar array regulation, battery charging and load voltage regulation for a suggested 3.3 volt DC power system. Multijunction solar array regulation with LiIon battery cell and voltage regulation to 3.3 volts need to be explored. The distribution effects of a 3.3 volt bus have been initially looked at, and would need further evaluation. Harness drops could be significant part of the voltage regulation parameters. LiIon charge parameters will be the dominate voltage swings, the worst case charge voltage needs be identified and assessed with the bus regulation requirements. LiIon discharge characteristics need to be addressed also for characteristics. Solar array and battery mode control needs to be identified with various charge control schemes. Circuit protection issues need to be addressed; are fuses or other circuit protection devices to be used. What energy, voltage and current will be needed to operate any circuit protection devices?

2. Power system requirements can be met with existing technology for the solar cells and battery cells, however vendor qualification programs are needed.

Solar Array Requirements. The Nanosat mission requirements show a 30.5 cm x 10 cm disc with a solar array area of 958 cm**2. Only half of this area will be in the sun at any one time and with the restriction of a 85% packing factor for solar cells, this gives a net usable area of 407 cm**2 or .0407 m**2. Taking into account the curvature of the disc, more that 7.38 watts are available from solar cells that would cover the parameter of the disc. A load analysis shows that this is ample power for the present mission requirements. Therefore no new technology efforts are required. A Qualification Program is recommended to qualify specific vendors and catalogue items for this mission.

Battery Requirements.

a. Survival power during eclipse. Assuming a passive thermal approach to keeping the spacecraft in survival mode during a 70 min eclipse, only two 1.2ah cells would be needed to meet a 1.7 Ah. If the spacecraft needs to operate through the eclipses, then additional cells may be needed. These cells presently exist and may require some modification, if any to make then space qualified. A Qualification Program is recommended to qualify specific vendors and catalogue items for this mission. This qualification program can be similar to the existing program, however with the cells specific to this mission. The expected cell weight would be 80 grams.

b. Full operation during eclipse. Assuming a passive thermal approach to keeping the spacecraft in operation mode during a 70 min eclipses, 12, 1.2 Ah cells would be needed to meet a 10 Ah requirement. The expected cell weight would be 480 grams. Different cell configurations will be considered to reduce the weight by going to a larger ah cell with lower number of cells.

3. Review of other alternate technologies.

a. Flywheel technology was looked at, however the limitation in the spacecraft size made it impractical. Also there were concerns with flying permanent magnets on a magnetometer mission. This technology was not looked at further for Nanosat.

b. Ultra capacitors were reviewed, however they would provide 4.5 Whr/l compared to 250 Whr/l for LiIon technology. Ultra capacitors have other attributes such as rapid discharge capabilities, however none of these other attributes are of benefit to Nanosat.

c.Single junction GaAs and Si solar cells were not looked at due to the limited array area and power needs for this spacecraft. A cost trade could be done in the future for single junction (SJ) GaAs vs Multi Junction (MJ) GaAs cells. This would depend on the load analysis at the time and the available solar array area vs the differential cost and performance parameters of the two cell types.

RF COMMUNICATION SUBSYSTEM

COMMUNICATIONS

The laws of nature place limits on how far we can push a given technology. Solar power available to a spacecraft is limited by the solar energy density and the size of the panels on the spacecraft. Solar cell efficiency can be increased from 20 % to 40 %, a factor of 2 improvement, but not by an order of magnitude to 200 %. For this, a new technology is required. RF communication data quality is limited by the noise background and the transmitted energy per bit (Eb). Once we get close to the physical limitations, we must change the technology to get more than an incremental improvement. For example, the communications coding proposal stated within this document is close to the Shannon limit.

Communication Requirements

1 Transmit science data to ground
2 Receive commands from ground
3 Generate signals required for orbit determination

Communication Derived Requirements

1a) Use communications coding to reduce power required for transmission
 b) Power density at Earth shall be at least $1.3 \times 10^{-17} w/m^2$ = -168.9 dBw/m^2 so that ground antenna need not be larger than 11m.
 c) Power density shall be less than …w/m^2/4KHz to be below RF flux density limitations.
 d) Data rate of approximately256 Kbps so that communication time is not unreasonable

2a) Sufficient solar array capacity for receiver constantly on.
 b) C & DH ability to decode and act on commands

3 Oscillator frequency stability when in sun light $< 5 \times 10^{-8}$/day so that one way doppler can be used for orbit determination.

Total Data and Communication Links

Table 1. below compares the communication time for several orbits and three data rates.

Table 1. Communication Time For Several Orbits and Data Rates

Apogee (Re)	Orbit Period (hours)	Data Rate Into Memory	Data per Orbit M bits	Data per Orbit M Bytes	Communication Time (min) 32 Kbps	128 Kbps	256 Kbps
12	29	2000	208.8	26.1	109.	27.	14
24	70	2000	504.	63.	263.	66.	33
42	150	2000	1080.	135.	563.	141.	70
60	249	2000	1792.8	224.1	934.	233.	117

Due to the low average power of about 0.5 watts = 500 mw available on the spacecraft for the communications subsystem, we expect to be able to send data to the earth only during the portion of the orbit near perigee, which is about 4.3 hours in duration. The range of the perigee portion is from 4.1 hrs to 4.3 hrs for the various orbits. In order to transmit to the ground at about 256 Kbps at X band, the link budget in Figure 1. shows that with an effective isotropic radiated power (EIRP) toward the earth of 500 mw = -3.0 dBw and an 11 m diameter ground antenna, a bit error rate (BER) of 1×10^{-5} can be established. We have been told by Jim Slavin that it is not necessary to take data while within 6.6 Re of the Earth. This will reduce the total data accumulated and reduce the time needed to dump one orbits worth of data.

In order to obtain an RF output of 0.5 watts, >2.0 watts of DC input power is required. Since the transmitter is only ON for about 1% of the orbit, batteries can be used to supplement the 0.5 watt allocation during transmission. For the large orbits, about 99% of the time the satellite will be in the sun so the charge rate can be 1% of the 2 watt use rate, hence 0.02 watts of solar cell power will provide for the transmitter. The remaining 0.48 watts is available for the receiver.

Battery size will depend on the worst case of apogee eclipse with instruments ON and transmitter OFF or perigee eclipse with instruments OFF but with transmitter ON.

Figure 1. Nanosat Downlink at X Band

1	Antenna gain (dB)	0		As much as 3 dB possible
1'	User EIRP (dBW)-1 ch.	-3.0		= 500 mw * antenna gain
2	Freq. (MHz)	8470		
3	Altitude (km)	32000		5re = 5*6387 = 31890
4	Ground elev. angle(deg)	90		straight up, Range = Alt
5	Range (km)	32000.00		
6	Space loss (dB)	200.99		
7	Atmos. atten (dB) [La]		1.40	80 F; 50 % RH
8	Rain atten (dB)[Lrain]		0.0	
9	Polarization loss (dB)		0.2	Circular polarization used
10	Pointing loss (dB)	0.2		
11	Multipath loss (dB)	0.10		S/C multipath
12	Rcvd power "den" (dBW)	-205.89		Line1-Sum(Line 6:Line 11)
13	Ground antenna size (m)	11		
14	Antenna efficiency (%)	55		
15	Antenna surf. tol. Loss (dB)	0.28		
16	Antenna gain (dB)	56.79		
16'	Rcvd Power (dBw)	-149.1		Line12+Line16, pwr at LNA input
17	LNA gain (dB)		50	high enough to define SNR
17'	LNA noise fig, N'(dB)		1	Tlna = (N-1)290 K 1 => 75
18	System temp (deg K)		125.09	Tb+Tlna, Tb=50 K
19	System temp (dB-K)	20.97		
20	System G/T (dB/K)	35.42		Line 16 - Line 19 - L
21	C/No (dB-Hz)		58.13	Line 12+Line20+228.6 k = -228.6 j/K
22	Data rate (bps)		256000	
23	Data rate (dB)		54.08	
24	Implementation loss(dB)	2.4		mod .6,diff codingloss.3,recv/BS =1.5
25	Avail Eb/No (dB)	2.11		Line 21 - Line 23 - Line 24
26	Eb/No required (dB)		1.85	10E-5, r = .45, k =24, BHD, 1.6+.25
27	Avail margin (dB)	-0.2		Line 25 - Line 26

Figure 2. Nanosat Uplink at X Band

	Antenna gain (dB)	56.79	Power = 10 watts = 10 dBw
1	User EIRP (dBW)-1 ch.	66.79	= 10w * antenna gain
2	Freq. (MHz)	7209.125	
3	Altitude (km)	32000	5re = 5*6387 = 31890
4	Ground elev. angle(deg)	90	straight up, Range = Alt
5	Range (km)	32000.00	
6	Space loss (dB)	200.99	
7	Atmos. atten (dB) [La]	0.40	80 F; 50 % RH
8	Rain atten (dB)[Lrain]	1.00	
9	Polarization loss (dB)	0.2	Circular polarization used
10	Pointing loss (dB)	0.2	
11	Multipath loss (dB)	0.10	S/C multipath
12	Rcvd power "den" (dBW)	-136.10	Line1-Sum(Line 6:Line 11)
13	Antenna size (m)	.07	Quadrafilar helix
14	Antenna efficiency (%)	na	
15	Antenna surf. tol. Loss (dB)	na	
16	Spacecraft Antenna gain (dB)	-5.0	
17	LNA gain (dB)	20	
17'	LNA noise fig, N'(dB)	3.5	Tlna = (N-1)290 K => 359.23deg K
18	System temp (deg K)	420.52	Tb+Tlna, Tb=61 K
19	System temp (dB-K)	26.24	
20	System G/T (dB/K)	-30.68	Line 16 - Line 19
21	C/No (dB-Hz)	61.82	Line 12 + Line 20 + 228.6
22	Data rate (bps)	1000	
23	Data rate (dB)	30.0	
24	Implementation loss(dB)	2.4	incl mod & diff coding loss
25	Avail Eb/No (dB)	28.86	Line 21 - Line 23 - Line 24
26	Eb/No required (dB)	13.5	10E-5, no coding, non coherent PSK
27	Avail margin (dB)	15.36	Line 25 - Line 26

Orbit Determination/Tracking

In the past, orbit determination was closely tied to the communication system for several reasons, today there are changes due to the Global Positioning System (GPS). With a spacecraft capable GPS receiver on board, a satellite can determine its position and velocity when it is within or near the GPS cluster. For Nanosat, GPS would supply measured data over only a small portion of the orbit and ground processing would still be necessary to project (propagate) the satellite position for every point of the orbit so that the measured science data can be correlated with the appropriate point in space. If GPS can be used, it is expected that it will give navigation information comparable to conventional range and doppler tracking. The current system and algorithms require that the signals from at least 4 satellites illuminate the user simultaneously in order for the user to obtain a navigation solution. This does not occur for a satellite above about 1000 Km from the earth's surface unless they can receive the weak signal from GPS satellites on the other side of the earth (that are not blocked by the earth). If a satellite has a stable oscillator and new algorithms it is possible to use the GPS data sequentially. Future GPS satellites may have a wider antenna beam to allow for users at a higher altitude; however, we are not aware of any serious planning in this direction prior to about 2010. GPS will require mass and power on board the satellite and its complication of both hardware and firmware (software) presents a degree of risk. Range and /or Doppler tracking involves a mass and power penalty but are less complicated and present less risk and appear to be a good solution for Nanosat orbits.

Ultra Stable Oscillator

If an ultra stable oscillator (USO) were used in the communication system, return only (one way) Doppler information could be used for tracking. A true USO, however, requires an oven and control electronics for accurate thermal stabilization. In the past it was unlikely this would fit the Nanosat mass and power environment. However recent development for Group Special Mobile (GSM, ~cell phone) and Personal Communication Systems (PCS) that are not currently space qualified must be considered. Reeves-Hoffman makes a 15 gm, low power oven controlled crystal oscillator in the 5 to 50 MHz frequency range that can achieve ± 5 ppb ($\pm 5 \times 10^{-9}$) over -40 _C to 85 _C. A preliminary study has shown that a true USO in not needed; an oscillator with a stability of 5×10^{-8}/day is sufficient to meet our orbit determination requirements of 20 Km.

For doppler only tracking, the doppler information is then used to determine the radial component of the spacecraft velocity and acceleration, which is then used to improve the orbit solution.

On Board Orbit Determination

Instead of using the telemetry signal from the USO on the spacecraft to do orbit determination on the ground, the uplink RF signal or a beacon from the ground can be used and compared to the USO on the spacecraft to do the orbit determination on board. An existing algorithm takes about 200 Kbytes of memory and about 0.2 MIPS (Cheryl Gramling 286 8002). This has been tested on a 1750 processor. A development effort is required to convert the algorithm to an 8 or 16 bit processor such as an AM186 which may be available in a small ultra low power format.

Communication Coding

Communication coding is a perfect example of a function that adds cost and complication to the ground station but is extremely beneficial to a satellite. Convolutional coding for example, is simple and adds very little mass or power requirement to the spacecraft but the commonly used code (r = 1/2, k = 7) enables the satellite-to-ground communication link to operate with about 1/4 of the power (-5.2dB) that would otherwise be required. The required decoding, (Viterbi decoding) adds complicated but not expensive circuitry to the ground receiver where it is easily tolerated.

A disadvantage of convolutional coding is that it causes the transmitted signal to occupy a greater RF bandwidth. The bandwidth expands inversely with the coding rate. Nanosats power is so low and the distance from earth at closest approach, perigee, is so large that the power spectral density will not be an environmental issue. As a matter of fact, the higher the coding rate, the lower the power per Hz.

High rate codes have the disadvantage that the data units, called convolutional symbols, that the receive system bit (symbol) synchronizer must synchronize on, contain only a fraction of the energy of each actual bit. At an Eb/No of .8 dB and a coding rate of 1/2 the symbol energy to noise ratio is half the as much, .8 dB - 3 dB = -2.2 dB. For a coding rate of 1/6, the energy is 1/6 as much, .8 dB - 7.8 dB = -7 dB. This results in a difficult and expensive symbol (bit) synchronization problem. We do not expect to use high rate convolutional codes for Nanosat.

Rate 1/2 turbo codes should be reevaluated as technology progresses, but for the "first cut", the rate 1/2 sequential convolutional code is being evaluated, with a bootstrap hybrid decoder (BHD) at the ground station. This software has been developed under a NASA grant to Daniel Costelo at the Indiana State University and is in our possession at the GSFC. This is a rate 1/2 code that delivers a $1x10^{-5}$ bit error rate at an Eb/No of 1.6 dB. The symbol energy to noise spectral density ratio is thus 1.6 dB - 3 dB = -1.4 dB. We believe this is a reasonable increment from current symbol synchronization equipment that will cost about $10,000. The software version of the BHD may not be able to perform in real time today, but probably will be on faster processors that will become available a few years from now.

Reed-Solomon Coding and Convolutional Interleave

Convolutional coding is particularly useful in a random or thermal noise environment as is found in space. For correction of burst errors, convolutional interleaving can be added after convolutional coding. Reed-Solomon coding can also be used to allow the system to correct errors in a noise environment that contains bursts. We do not expect the Nanosat ground antenna to be looking into a bursty environment except during a moon or sun intrusion into the ground antennas field of view. Sun intrusion will be an issue twice a year. This needs to be examined along with the ground station scheduling simulation. As part of the Ultra Low Power technology development, Reed-Solomon chips are currently being tested at GSFC and if available will likely be used on Nanosat.

Consultative Committee for Space Data Systems (CCSDS)

The CCSDS has made a wide range of recommendations in an effort to standardize spacecraft communications. At the highest level, these recommendations allow ground stations to receive and forward data for multiple missions. By adhering to a standard RF and frame structure, space agencies of various nations can share spacecraft support requirements and automatically pass the data to the cognizant institution. The full CCSDS frame and packet structure is designed for large spacecraft with multiple instruments. The data from each instrument is placed on a different virtual channel and given a virtual channel identification number (VCID). Within a given instrument, various packets are constructed containing data that has various applications within that instrument and are identified by their application ID, APID. Fill frames are generated as needed to maintain a constant bit rate and they are given a VCID of 63.

The Nanosat data system should be design for a minimum of the CCSDS structure and should not waste telemetry bandwidth on fill frames. The science and housekeeping frames should have a predetermined sequence that can be adjusted by ground command (unless autonomous) but should remain within the commanded data rate. The standard 32 bit CCSDS frame marker should be used and the PN transition generator should be used. This is required to ensure good receiver and bit synchronizer lock. The randomizing effect of the common $r = 1/2$, $k = 7$ convolutional code can not be relied on since regions of data with all 1s or all 0s may not be sufficiently randomized. Inverting several of the code generators in the proposed sequential code may be sufficient to randomize strings of 1s or 0s in the data. Until this is determined, the rule of thumb is use the CCSDS randomizer also. It is likely that the science data will be compressed and the convolutional decoder (BHD) output of 1×10^{-5} will not be sufficient, hence, an outer Reed-Solomon will probably be required.

Ground System

If the on board memory is sufficient to hold only one orbit worth of data for the largest orbit, then satellites in that orbit must be treated with the highest priority as they pass close to the Earth.

THERMAL SUBSYSTEM

Technical Discussion

The Nanosat preliminary thermal design is primarily a passive design consisting of multilayer insulating blankets (MLI) and selected surface-finish applications. The MLI will be used to reduce losses to and gains from the environment. This preliminary passive design philosophy incorporates a low cost approach to maintaining temperature requirements throughout the Nanosat payload.

Preliminary Study

A simple geometric math model representing one Nanosat was created using a basic cylindrical shape with a 30 cm diameter and 10 cm height weighing about 10 kg. Preliminary properties were picked based on typical optical properties of common coatings. Internal and external radiation couplings were calculated using SSPTA. Hand calculations were done to determine the total energy of the sun incident on the solar cells of the spacecraft.

A simple thermal math model was created using SINDA85. The model has 11 nodes, 10 for the Nanosat spacecraft and a boundary space node. The internal conductors, including conduction and radiation, were calculated using best engineering judgment.
The effective emittance range (e*) for the MLI blankets was varied from 0.02 $\pm$ 0.01. The total internal power dissipation of the spacecraft was run using two different powers of 3 and 6 watts, respectively, during the operational portion of the orbit and zero watts during the eclipse portion. To date, it has been determined that there will be no heater power available during the eclipse portion of the orbit.

The conceptual thermal design approach of the Nanosat satellite entailed three configurations. The objective of the analysis was to develop a thermal design to investigate the characteristics and potential benefits of each of the design strategies. The first configuration (2a) insulated the top and bottom of the spacecraft only. For the second configuration (2b), the entire spacecraft was insulated except for a small radiator sized to handle the internal dissipation during the sunlit mission phases. For the third configuration (2c), the entire spacecraft is insulated and a mini-CPL (capillary pumped loop) was modeled to transport the internal heat to an outer radiator. All three configurations assumed that the solar cells were isolated from the spacecraft top and bottom.

Each of the three configurations included optical properties shown on the following pages.

Configuration 2a.
Insulation on only top and bottom

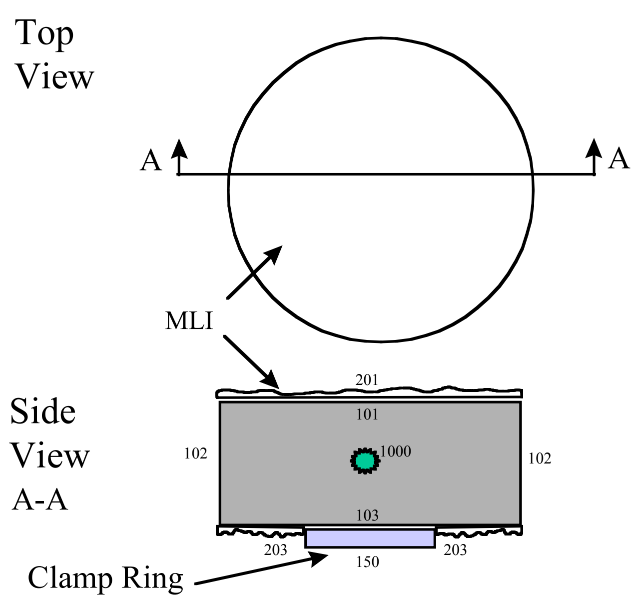


Figure 1. Configuration 2a: Top and Side View

Table 1. Configuration 2a: Preliminary Thermal Optical Properties

Node	Description	Coating	alpha	emis	e*
101	Spacecraft top (internal)	Black	0.90	0.90	

102	Solar cells & S/C core	GaAs solar cells	0.90	0.77	
103	Spacecraft bottom (internal)	Black	0.90	0.90	
150	Clamp Ring	Irridited aluminum	0.25	0.11	
201	MLI on top of S/C	5 mil Kapton	0.49	0.83	0.02 $\pm$ 0.01
203	MLI on bottom of S/C	5 mil Kapton	0.49	0.83	0.02 $\pm$ 0.01
1000	Internal lump node	Black	0.90	0.90	

Configuration 2b.
Entire spacecraft insulated except for small radiator

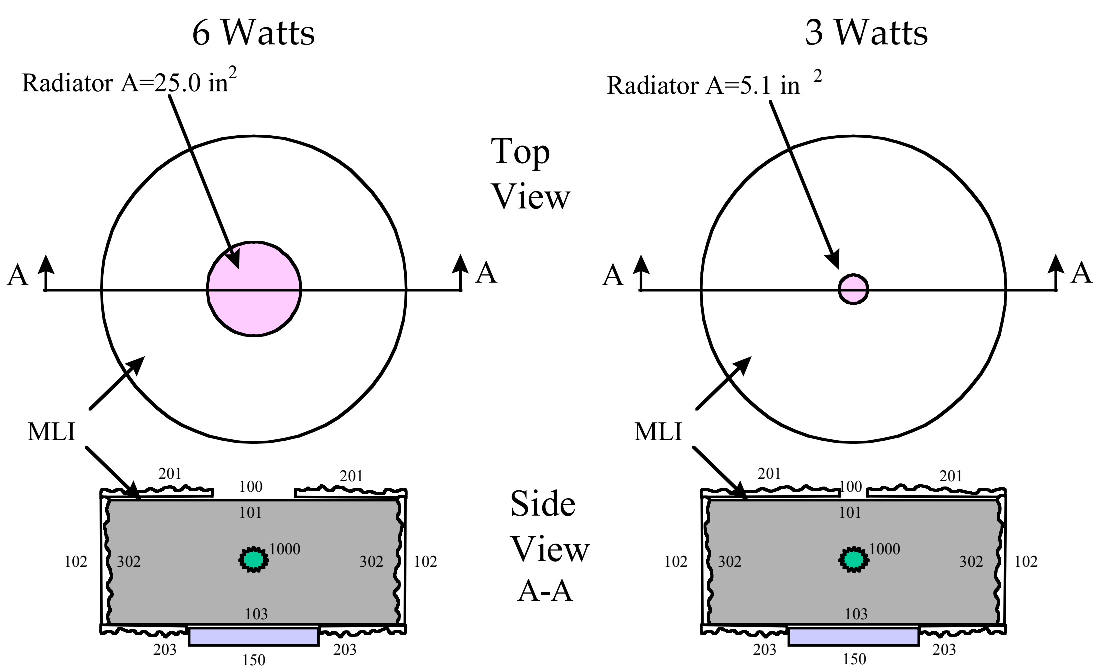


Figure 2. Configuration 2b: Top and Side View

Table 2. Configuration 2b: Preliminary Thermal Optical Properties

Node	Description	Coating	alpha	emis	e*
100	Spacecraft radiator	Silver Teflon	0.08	0.75	
101	Spacecraft top (internal)	Black	0.90	0.90	
102	Solar cells & S/C core	GaAs solar cells	0.90	0.77	
103	Spacecraft bottom (internal)	Black	0.90	0.90	
150	Clamp Ring	Irridited aluminum	0.25	0.11	
201	MLI on top of S/C	5 mil Kapton	0.49	0.83	0.02 ± 0.01
203	MLI on bottom of S/C	5 mil Kapton	0.49	0.83	0.02 ± 0.01
302	MLI internal of solar cells	2 mil Kapton	0.39	0.73	0.02 ± 0.01
1000	Internal lump node	Black	0.90	0.90	

Configuration 2c.
Entire Spacecraft insulated
CPL between radiator and internal power dissipation node

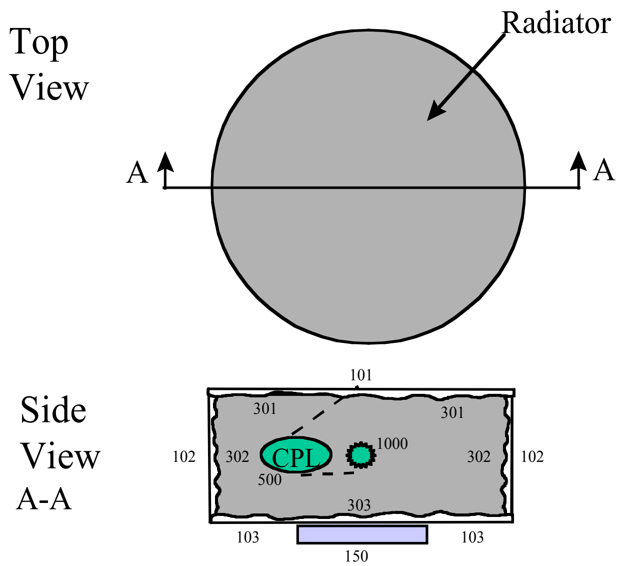


Figure 3. Configuration 2c: Top and Side View

Table 3. Configuration 2c: Preliminary Thermal Optical Properties

Node	Description	Coating	alpha	emis	e*
101	Spacecraft top (external) radiator	Silver Teflon	0.08	0.75	
102	Solar cells & S/C core	GaAs solar cells	0.90	0.77	
103	Spacecraft bottom (external)	Irridited aluminum	0.25	0.11	
150	Clamp Ring	Irridited aluminum	0.25	0.11	
301	MLI on inside top of S/C	2 mil Kapton	0.39	0.73	0.02 ± 0.01
302	MLI internal of solar cells	2 mil Kapton	0.39	0.73	0.02 ± 0.01
303	MLI on inside bottom of S/C	2 mil Kapton	0.39	0.73	0.02 ± 0.01
500	CPL connects lump 101 to 1000	---	----	---	
1000	Internal lump node	Black	0.90	0.90	

Preliminary mini-CPL requirements

Background

Capillary Pumped Loop (CPL) systems have been under development at GSFC since the mid-1980's. They utilize two-phase ammonia as the working fluid with wick materials used to provide the pumping action. CPL systems provide constant temperature heat sinks for wide ranges of power input and transport distances. The Earth Observing System (EOS-AM) has baselined CPL's for the thermal control of three of its' instruments. These systems are relatively large, generally utilizing 1 inch diameter pumps and providing transport capabilities of several hundred watts.

The recent emphasis on the implementation of smaller satellites leads to a requirement for development of smaller subsystems in several areas. These newer systems (such as electronics boxes and instruments) often have higher power densities and thermal transport requirements than seen on previous small satellites. Older thermal technology, such as heaters, thermostats, and heat pipes, may not be sufficient to meet the requirements of these new systems. Heat pipes have limited transport capability and require additional heater power to provide precise temperature control.

Objectives of a mini – CPL breadboard

The goal of the project is to design, build, and test a miniature breadboard CPL system that would demonstrate the feasibility of a small CPL. This system, CPLjr, would have power capabilities in the range of 10 watts or less, but would still maintain the temperature control and transport capabilities of current larger systems. CPLjr would provide thermal control over typical spacecraft ranges. Scaling issues related to the down-sizing of the loop components will be addressed. Recent work on cryogenic CPL's has shown promise in the miniaturization of CPL components.

CPLjr Schedule:

Define Requirements	3 months
Loop Design	3 months
Fabrication	6 months
Testing	6 months
Final Report	
Total Anticipated Schedule:	18 months

Preliminary CPL Modeling

In configuration 2c, a mini-CPL was modeled very simply. It was assumed that the capillary pumped loop system would remain at a constant temperature of 20°C and would transport the internal heat to an outer "skin" radiator during operations and "shut down" during the eclipse to maintain temperatures as warm as possible. In order to meet the Nanosat preliminary thermal requirements as shown below in Table 5, the following miniature two-phase device options were identified: Loop Heat Pipes (LHP) and Capillary Pumped Loops (CPL).

Table 5. Nanosat Preliminary Design Requirements

Heat Loads	2 W to 10 W
Maximum non-operating environmental temperature	60 °C
Operating temperature range	0-50 °C, 20°C nominal
Eclipse/Shutdown Minimum instrument temperature Minimum sink temperature Duration	 -20°C -60° C 475 minutes
External Body Forces Ground testing Centrifugal acceleration (S/C only)	10 cm can height -> 0.6 kPa for Ammonia -> $r_{p,min}$=60 micron wick 10-20 RPM with 15 cm radius -> .02 m/s^2
Reliability	single fault tolerance assumed (requirement TBD)
Mission Life	1 year

Table 6. LHP/CPL Design

Materials	LHP/CPL: Stainless Steel transport lines, metal or polyethylene wick material, aluminum housing, Inconel condenser tubing and transport lines if temperatures drops below -60°C. Note: Polyethylene wicks can not be used if the evaporator temperatures drop below -20°C.
Equipment interface	CPL: 1 inch x 1 inch - flat plate interface
Direct condensation radiator interface	CPL: 1/16" Inconel tubing, 700 cm^2 radiator area, <10 cm condenser length

Weight	CPL: < 0.3 kg
Transport length	CPL: 0.0625 in diameter tubing with 0.01 in wall; 10 cm vapor line; 10 cm liquid line; longer transport lines are easily accommodate
Reverse conductance	CPL: 3.5×10^{-4} W/K (for two 1/16 OD Stainless tubes with 0.01 wall)
Design pressure	CPL: Ammonia at 60°C with 1.5 FS = 3900 kPa (570 psi)

Note: A TAG-3 heat pipe extrusion (0.375 inch OD) can be used for the LHP or CPL evaporator. Based on recent calculations, a flat plate evaporator design with a stainless steel wick may also be a viable option with the LHP.

Table 7. Design Issues or Concerns

Transport length and ground testing limits	For metal wicks (r_p=2 micron, perm=3×10^{-14} m^{2}) pumping head is 2 meters and for porous polyethylene wicks (r_p=15 micron, perm = 1.5×10^{-12} m^2) the pumping head is 40 cm. At the low flow rates associated with 10W, transport line length and ground testing limits are not an issue.
Temperature control	LHP: Tight temperature control is not available; the operating temperature will adjust according to the sink temperature and heat load (see table 5). CPL: Precise temperature control requires a cold biased reservoir and heater power to properly maintain the reservoir temperature at the set point
Start-up prep time	LHP: None CPL: Pressure prime is required to condense vapor in the evaporator liquid core. The reservoir temperature should be elevated 5°C above the evaporator temperature and maintained for approximately 30 minutes prior to applying heat to the evaporator. Reservoir

	heater power and actual prep time TBD.
Minimum heat load 2 W	LHP: Not a problem, the LHP will adjust the operating temperature according to the power input and sink temperature (see table 5). CPL: If the CPL reservoir temperature is regulated, the temperature liquid at the evaporator inlet will increase because of the low flow rate; this may cause the liquid to boil in the evaporator core and possibly deprime the system.
Eclipse - Power requirements	LHP: No power required CPL: Reservoir heaters or PCM heat exchangers are required to maintain system prime if quick start-ups are required.
Eclipse - Minimum instrument temperature	For the design reverse conductance of 3.5×10^{-4} W/K (see Table 6), the thermal mass of the instrument and LHP/CPL system must be greater than 15 joule/K to meet requirements of a 475 minute eclipse with radiator temperatures at -60°C and still maintain instrument temperatures above -20°C. The LHP/CPL TAG 3 evaporator thermal mass with aluminum housing and 1/8 inch thick heat acquisition saddle and stainless steel wick is 8.23 joule/K, therefore maintaining instrument temperatures should not be an issue as long as the instrument thermal mass is greater than 6.8 joule/K. An additional 7.7g of aluminum (or 14g of stainless steel) can be added to the evaporator interface plate (see Table 6) to compensate if the instrument thermal mass is below 6.8 joule/K
Freeze recovery	LHP/CPL: Post-eclipse heaters to thaw condenser and transport lines would be sized at 1 watt to thaw 0.25 grams of ammonia (assuming freezing is restricted to only the transport lines and condenser) in less than two minutes.
Shut down to minimize reverse conductance	LHP/CPL: Option 1 - Heater activated shutdown: Prior to shut down, quickly increase reservoir temperature 5°C above evaporator temperature (Time and power TBD); design system so

	reservoir temperature remains above evaporator temperature (i.e. a cold biased CPL reservoir is not an option without a reservoir heater or PCM heat exchanger to maintain temperature during the eclipse) Option 2 - <u>Variable volume reservoir</u>: Use bellows to vary the reservoir volume and change the pressure inside the reservoir. Option 3 - <u>Reservoir thermal mass/PCM</u> Option 4 -<u>Differential expansion vapor line choking mechanism</u>: This would block the fluid flow and prevent fluid circulation during the eclipse. Option 5 - <u>NCG expansion shutdown</u>: This would fill the evaporator with vapor to stop pumping (not an option for CPL)
Technology maturity	<u>LHP/CPL</u>: Further development required. Shutdown options 2 through 5 require development and testing. Design Heritage: *<u>LHP</u>: GSFC Glove LHP (recently developed) and miniaturized ALPHA (to be developed) *<u>CPL</u>: Cryo-CPL (developed)

<u>Preliminary Results</u>

The tables on the following three pages show temperature ranges for each of the three configurations assumed. The results presented indicate basic important features of each design strategy and help to bound a thermal design, but do not reflect individual component temperatures due to the simplified nature of the model.

For configuration 2a, where only the top and bottom are insulated, the in sun (operational) temperatures are shown in the table at about 30 to 35°C. This level could easily be adjusted to near room temperature with a relatively small radiator area on the sides or end panels of the spacecraft. The key advantage of this configuration is its reliability, or robustness. Since the temperature of the spacecraft is set by a high energy balance (heat in = heat out) dominated by the solar energy absorbed by, and energy radiated from the side solar arrays, the operational temperature of the spacecraft is insensitive to top & bottom MLI properties or internal heat dissipation. This can be seen by the relatively small variation of the internal lump node temperature as the multilayer insulation (MLI) effectiveness (e*) and internal heat dissipation is varied (3 watts, 6 watts). However, the feature that yields the operational reliability also results in a rapid drop in temperature when the solar load disappears during the eclipse. At end of the 475 minute (almost 8 hour) eclipse, internal temperatures drop by about 60°C (stated as a drop since operational temperatures have not been optimized for the 3 configurations studied), resulting in temperatures in the range of -30 to -40°C. At the same time, the solar arrays have dropped to a temperature of about -60°C. Since specific equipment with specific temperature limits have not been identified to date, the feasibility of these end-of-eclipse temperatures must be judged on the basis of general spacecraft equipment. Based on past experience, these end-of-eclipse temperatures are reasonable, at least as survival temperatures, for at least some spacecraft electronics. Certain other components may have a problem with these temperatures. End-of-eclipse solar array temperatures are not a problem. Based on this, and given the inherent reliability of this approach, configuration 2a should be retained for further consideration.

Configuration 2b, which is fully insulated except for a sized passive radiator, would result in operational temperatures about the same as configuration 2a. Because of the insulated nature of the design, with a much smaller overall energy balance than configuration 2a, this configuration is much more sensitive to MLI properties, and internal power dissipation than configuration 2a. Although the temperatures presented in the table have not been optimized, this sensitivity can be seen in the variation of the internal lump node temperatures presented as a function of the MLI effectiveness (e*), and the resizing of the radiator required for the 2 power conditions analyzed, shown in Figure 2. However, eclipse performance improves. During the ~8 hour eclipse, internal temperatures drop by only about 20°C, a marked improvement, with end-of-eclipse temperatures well within the range of most spacecraft components. It should be noted that the solar arrays, since they are now isolated from the body of the spacecraft, drop to temperatures of about -110°C. Even these solar array temperatures should not pose a problem since the solar arrays of many geosynchronous satellites drop routinely to temperatures of about -150°C during the 72-minute eclipse experienced by these spacecraft at each equinox season. So, if warmer end-of-eclipse equipment temperatures are required, they can be achieved, at the expense of some reliability, with this "passive" thermal design.

The key feature of configuration 2c is that the internal equipment is completely isolated, both radiatively and conductively, from the outside "shell" , and the equipment is coupled to an external radiator with the capillary pumped loop (CPL) or loop heat pipe(LHP). Operational temperatures are maintained to temperatures of about 20°C nominal, with the temperature totally dependent on the operation of the two-phase "loop". The two-phase heat transport can be made redundant by the addition of a second loop if single fault tolerance is desired (note that this is not a consideration for configurations 2a and 2b). During the ~8 hour eclipse, a further improvement is realized, with internal temperatures dropping by as little as 6°C if the internal payload is WELL insulated from the exterior of the spacecraft. Note also the sensitivity of internal temperatures to the MLI effectiveness and whether the "payload" (internal node) is conductively coupled or isolated from the bottom of the spacecraft. As in configuration 2b, the solar array temperatures drop to about -110°C. For certain equipment or instrumentation, the temperature control afforded by this type of "active" design may be necessary, so consideration of this design should also be continued. Note that it has been assumed in this study that no power would be available during the eclipse to keep the loop "shut down". Since, currently, utilization of a small amount of power is used to "shut down" two-phase systems, this would constitute a technology development area, in addition to the small heat transport requirements of a Nanosat.

For all configurations, the resultant temperatures for the eclipse portion of the orbit are a function of time in the eclipse and mass distribution assumed.

The results of this study, to date, help to indicate a direction the thermal design should take as details evolve. The thermal design presented here is a first-cut in the Nanosat feasibility study. It may not reflect the actual configuration of the Nanosat and is based on thermal design guidelines available at the time. The model will later be refined to support any systems design tradeoff studies and detailed design as required.

Future

The development of a "mini-CPL" has been identified as a candidate for technology development funds. A tentative plan has been formulated for this technology development effort.

Areas of further analytical study include the following, as details of the Nanosat configuration are further developed:

- "Nano" electronics packaging
- "Nano" propulsion system requirements
- Effects of magnetometer boom and other "realities" on the above study results

FUTURE WORK

1.In general (over the long term), completion of design, development, integration, test, launch and mission operations.

2.Technology Research & Development (examples in progress)

A. C & DH packaging
B. Miniature earth & sun sensors
C. Miniature propulsion components
D. LiIon battery cell development
E. GaAs triple junction solar cells
F. Low voltage bus topology work (3.3 volts)
G. Structural battery
H. Hybrid electronics battery
I. Ultra Low Power Radiation Tolerant Electronics (CULPRiT)
J. Light weight low cost structures
K. Deployership release mechanisms
L. High conductivity diamond face sheets
M. Mini heat transport systems (CPL & LHP)
N. Variable emittance coatings (MEMS louvers)
O. Miniature low voltage high efficiency x-band transmitter
P. Comprehensive investigation of applicability of present technology R & D to Nanosats

3. Studies and analyses over the short term:

Firm up the preliminary analyses presented in this document:

A. Mission orbits
B. Operations concept
C. Mission instrument details
D. Nanosat and Deployership system and subsystem details

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References

P. V. Panetta, H. C. Culver, J. Gagosian, M. Johnson, J. Kellogg, D. Mangus, T. Michalek, V. Sank, S. Tompkins: 1998,
NASA-GSFC Nano-Satellite Technology Development,
12th AIAA/USU Conference on Small Satellites (SSC98-VI-5)

M. Concha, R. DeFazio: 1998,
NANOSATCONSTELLATION MISSION DESIGN,
Proceedings of the American Astronomical Society,
GSFC/International Symposium on Space Flight Dynamics 1998,
AAS 98-305

K. A. Crane (Swales Aerospace) to T. Michalek (NASA/GSFC),
September 30, 1997,
Nanosat Feasibility Study Summary Report,